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# REPORT

NASA CR-473

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# INTEGRATED SPACECRAFT
# DISPLAY DEVELOPMENT

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

# FOREWORD

This Final Report on the Integrated Spacecraft Display Development study has been prepared for the Flight Research Center, National Aeronautics and Space Administration, Edwards, California, by the Information and Communications Division of Sperry Gyroscope Company, Division of Sperry Rand Corporation, Carle Place, New York, under Contract NAS 4-385.

This report was prepared by the Windshield Display Group of the Information and Communications Division of Sperry with assistance from the Sperry Rand Systems Group and the Sperry Flight Research Group.

The objectives of the study were to provide information regarding the application of projected display technique principles, previously developed and flight tested by Sperry in aircraft, to manned spacecraft use, and to provide information which permits the establishment of a complete analog simulator evaluation program for this display technique.

# ABSTRACT

A study was made of the application of projected display techniques to manned space flight.

The study consisted of two specific tasks:

1. Generation of a design specification for a projection display system suitable for use in a fixed-base spacecraft simulator.

2. A preliminary design investigation of the use of projection display techniques in spacecraft was conducted which included consideration of such factors as information content, projection display/panel integration and display/control compatibility. Trade-off studies of various electronic and optical systems were included.

The use of the display system was considered for attitude and translational control for rendezvous and lunar landing. Information was generated to permit the establishment of a complete analog simulator evaluation program for this display technique.

# SECTION I

## INTRODUCTION

### 1-1. GENERAL

This report of the study of Integrated Spacecraft Display Development has been prepared for the Flight Research Center, National Aeronautics and Space Administration, under contract NAS 4-385, by the Information and Communication Division, Sperry Gyroscope Company, Division of the Sperry Rand Corporation. The objective of this study has been, first, to provide information so that the applicability of the projected display technique to spacecraft can be determined and, second, to provide information which will permit the establishment of an analogue simulator program for this display technique.

Recent instrumentation developments in the aircraft field have led to the concept of "Heads-Up", or windshield displays, which permit the pilot to view the outside world simultaneously with instrument displays. Of particular application to this study are displays which make use of optical projection techniques, similar to gun sights, which permit the superpositioning of images, defined by collimated light rays, on the external visual field. Such displays are free from parallax and appear to be identified with the background in three-dimensional space.

Heads-up display development programs at Sperry in the aircraft field have been based on the concept of using the VFR (visual) piloting skills in the IFR (instrument) situation. This has been accomplished by generating displays which enhance the visual flight skills and provide display elements which are compatible in position and motion with the natural earth elements. Through this means the same piloting skills can be used in either the VFR or IFR environment. Programs conducted for the Air Force and Navy in jet aircraft (references 2 and 3) have proven the effectiveness of this display concept in furnishing precise flight path and speed control information during the difficult approach and landing maneuvers.

This study has considered the application of this compatible VFR/IFR heads-up display concept to manned spacecraft. Primary attention has been paid to the control and assessment tasks anticipated for rendezvous and lunar landing operations. It appears likely that this form of instrumentation is feasible for spacecraft installation and should be fruitful in increasing the contributions of man in the spacecraft mission.

## 1-2. EXPLOITING MAN'S SKILLS IN SPACECRAFT CONTROL

The following visual behavioral capacities of the astronaut form a basis for designing displays which will augment manual control and assessment of spaceflight operations.

- Orientation
- Visual perception in space
- Visual discrimination
- Visual guidance.

### A. ORIENTATION

Spaceflight provides the observer with the novel situation in which his environment can be viewed from any possible geometric orientation for extended periods of time. Furthermore, the vehicle orientation is under the complete control of the astronaut and the attitude of the vehicle has no relationship to the direction of its motion. This situation represents a radical departure from even the most unusual operations within the atmosphere such as aerial acrobatics. It becomes truly unique when the loss of some of the subjective orientation cues due to weightlessness is considered.

The human observer generally receives two types of cues to orientation, vestibular and visual. The absence of the vestibular cues in the weightless condition, which is characteristic of spaceflight operations, enhances the importance of visual cues. The vestibular cues normally provide the human with an "upright" orientation. For example, in aircraft operations a pilot using "outside-in" type of flight instrument displays senses the apparent "gravity" in the aircraft which he can interpret as being parallel with the actual gravity field. His orientation in this case is related to the earth. This same type of panel display philosophy may be seriously deficient if applied to spaceflight. This appears likely since the vestibular cues (which make the pilot's subjective orientation "believable") have been removed. On the other hand, visual cues to the "real world" are available almost without limitation in space. There is no "weather", and obstructions to vision will be man-made. Accustomed from early childhood to rely on visual information, our conception of the world is basically visual (not auditory, not olfactory, etc.). Many examples in the sports world point to the ability of athletes to maintain orientation in a visual world while in a "near weightless" condition. Sky diving, springboard and platform diving, and trapeze acrobatics are a few examples.

Display techniques which augment the visual cues received by the astronaut may facilitate orientation in space. However, these augmenting

cues must be compatible with visual perception in space for the dividend to accrue. Panel mounted instruments are of limited utility in this regard since their indications must be assimilated and then correlated with the external visual field by the astronaut. Optical projection techniques in which there is a superposition of reference indications on the external real world provide a higher order of phenomenal organization of these elements in the visual field. The application of these techniques to spacecraft would therefore strengthen the astronaut's perception of himself in relation to the space in which he is controlling his vehicle.

## B. VISUAL PERCEPTION IN SPACE

The visual environment in space is substantially different from the environment that is experienced within the atmosphere. These differences may have adverse effects upon the astronaut's ability to perform visual control tasks. One of the differences is the intense contrast between illuminated and unilluminated surfaces. It is estimated that an earth satellite's illuminated surface due to the sun's direct radiation and reflected light from the earth would have a luminance in the order of 5, 000 to 10, 000 foot-lamberts. This is approximately the luminance of a snow field on a bright clear day. This intense visual stimulus, if viewed against a background of stars, may influence the ability of the astronaut to perceive movement of the satellite. Reduction of apparent intensity by filters has the effect of reducing the visibility of the background of stars, thus introducing a different object-background relationship.

The two perception problems in the rendezvous in space are as follows:

- The visual search and acquisition of target

- The perception of range and relative movement.

The first problem can be formidable since relatively small windows can be afforded, at least for re-entry vehicles. The astronaut must orient the vehicle in the necessary direction through instrumentation and then scan the field of view to locate the target. When on considers that the target must be perceived against an extreme range of possible backgrounds, it is likely that unaided visual search may not permit early acquisition. Relative angular movement of the target will require changes in the spacecraft orientation during the search process and further complicate the search operation.

Another unusual aspect of visual spaceflight occurs during the final closing operation of the rendezvous prior to docking. The visual

control of the final closing requires that the astronaut establish the translational motion of his spacecraft relative to the target. This can only be done if the astronaut first stops his rotation with respect to a stable frame of reference which can be seen. It appears likely that only the horizon and the stellar field can meet this stability requirement. The earth below the spacecraft is moving angularly at a rate of approximately 170 degrees per minute for an orbital altitude of 100 miles. This angular rate precludes its use as a stable background. When looking up at the target against a star background, the astronaut can maneuver to maintain the target against this background. However, this operation may not be as efficient as the case where the target is maintained in a fixed position relative to the local vertical. To perform the latter operation, the visual field must be augmented with instrumentation.

## C. VISUAL DISCRIMINATION

The human observer can detect and discriminate details separated by a few minutes of visual angle. The exact values which can be utilized in practice depend on the viewing conditions. Guidance and flight control information can be presented to the astronaut with this high order of resolution through optical projection techniques. Panel mounted instruments may also provide a high degree of precision, but only with higher sensitivities. These sensitivity requirements compromise compatibility with the real world situation, or require unusual size which conflicts with practical installation in a space vehicle.

## D. VISUAL GUIDANCE

The astronaut can visually perceive a range of objects and regions of interest in space which cannot easily be matched by any known system of instrumentation, irrespective of complexity. Some examples are: a rendezvous target over a range of distances from final approach to the docking maneuver, including its size and attitude with respect to the approaching spacecraft; an appropriate landing site on the lunar surface; obstacles in the intended landing path of the spacecraft on the moon. The perception of these items provides the basis for the visual guidance of the spacecraft for short term control maneuvers.

When guidance is obtained visually, it is natural to consider augmenting the visual flight control capability of the astronaut by projecting information regarding the flight path of the vehicle on the same external visual field. The relative direction of translation (aiming point) between the two bodies is one such parameter for display. Relative transverse

motion between the spacecraft and the target is immediately perceived as a separation between the aiming point image and the target. This separation visually defines the miss distance. The superposition of aiming point and target indicates zero transverse range rate, i.e., a collision course.

Command information to achieve velocity increments in any direction may also be presented by projected images. This enables the astronaut to control the spacecraft and monitor the results of his control actions without shifting from visual reference to the outside environment.

When both visual guidance and flight control information are in the same field, a novel opportunity for "quickening" the control situation exists. Tracking a target can produce controlled motion along a selected approach path to the target. The source of guidance in this situation is the astronaut himself, through visual perception of the target.

## 1-3. PROGRAM

The program implemented to satisfy the NASA project objectives had two aspects. The first was the specification of a projection display system suitable for use in a fixed base spacecraft simulator. The second task was a comprehensive preliminary design investigation to furnish NASA with data to permit establishing a preliminary design specification for a spacecraft projection display unit, and to form a basis for a detailed analog simulator evaluation program.

The design investigation covered rendezvous and lunar landing mission definitions and the pilot information requirements for manned control and assessment of the maneuvers. The information requirements were translated into heads-up display image configurations based on specific criteria for the allocation of information to the projection display and the control tasks required.

Integration of panel and projection displays and display control compatibility were studied.

Electronic system trade-off studies compared size, weight, power and reliability for three levels of display comprehensiveness and complexity. Basic design and application data for reflection and refraction optical systems has been generated to simplify this difficult trade-off task.

An investigation of the feasibility of incorporating display images into an overlapping-monocular viewing system is the subject of an amendment to the basic contract. This work is not described in this report. It will be the subject of a separate report at the completion of the additional work task.

## 1-4. REPORT ORGANIZATION

This report includes all the significant results of the studies conducted on the program. The simulator display system specification is described first, followed by a detailed treatment of the many aspects of the preliminary design investigation. A number of illustrations are included to show the various display systems and to present the trade-off study results.

# SECTION 2

## DESIGN SPECIFICATION FOR AN ANALOG SIMULATOR
## DISPLAY SYSTEM, TASK 1.03A

### 2-1. DESCRIPTION OF TASK AND PROGRAM PLAN

#### A. TASK DESCRIPTION

The purpose of this task is to provide NASA-FRC with adequate information to specify and procure a flexible projection display unit that will be compatible with a planned visual simulator. The detailed requirements for the specification are:

- General description
- Description of display system outputs
    - Basic forms, capacity, response and accuracy
    - Recommended display configuration capability for the space flight modes
- Specification for operating environment and packaging form factors including size and weight
- Specifications for input requirements to the display system
- Specifications for interface requirements for adapting the display system to the analog computers designated by NASA-FRC
- This specification will define a projection display system for use in a fixed base spacecraft analog simulator

#### B. PROGRAM PLAN

The following studies were defined and conducted to generate the specification.

##### 1. Display System Output Study

Generate the requirements for display system outputs consistent with:

a. The suggested display cues of the original proposal on the program

b. Further study of manned space flight techniques and display/control methods for them

c. Subsequent consultation with and guidance of NASA-FRC

d. The requirements for sufficient capacity and flexibility of cue forms to be useful in an experimental analog simulator program

e. The display capability of the design of the existing projection display equipment.

2. Input and Interface Requirements Study

Generate the input and interface requirements on the equipment considering:

a. Required display outputs

b. Input-output characteristics of the analog simulator computers designated by NASA-FRC

c. Input-output characteristics of the design of the existing projection display equipment

d. Requirement for sufficient flexibility to be useful in an experimental analog simulation program.

3. Environment and Packaging Form Factor Study

Generate requirements on equipment considering:

a. Operating environment

b. Available power sources

c. Space available for the optical sighthead of the projection display in both the rendezvous and lunar landing configurations of the spacecraft simulator

d. Space and form factor for electronics to be compatible with other NASA-FRC laboratory equipments.

C. DISCUSSION OF RESULTS

The results of the study were the generation and deliver to NASA-FRC of the "Design Specification for an Analog Simulator Display System" (Sperry document Specification No. 9202-1002 dated September 5, 1963). The specification is included as Appendix A of this report and only brief general comments will be included in this section.

The specification defines the technical requirements for a display system for use in a fixed base spacecraft analog simulator. The display system presents to the astronaut various forms of flight control and assessment information on a window so that the information is superimposed on the simulated real-world display of the spacecraft simulator.

The display information, where desirable, is compatible, on a one-to-one relationship, with a simulated real-world display. The display information consists of artificially generated images focused at the same screen image plane as the simulated real-world display. The images are generated by a cathode ray tube and an illuminated reticle.

The display system has suitable capacity and flexibility of display information for experimental use in the spacecraft simulator.

The equipment is usable with fixed base spacecraft simulators with analog computers of the 31R, 131R and 231R types.

The equipment consists of a Sighthead Unit and two Electronics Units. The Electronics Units are lab-rack type units for conformity with other simulator equipments and for ease in handling and maintenance.

The display equipment, with minor modification, is suitable for use in overhead or instrument panel locations.

# SECTION III

## PRELIMINARY DESIGN INVESTIGATION, TASK 1.03B

### 3-1. DESCRIPTION OF TASK AND PROGRAM PLAN

#### A. OBJECTIVE

The object of the preliminary design investigation is to provide information to permit the determination of the applicability of the projection display technique to spacecraft, and information which permits the establishment of a complete analog simulator evaluation program for this display technique.

#### B. PROGRAM PLAN

The objectives of the subject task have been satisfied through the study program briefly described below.

##### 1. Mission Definition and Analysis Study

Detailed definitions were established for the specific mission phases (rendezvous and lunar landing) specified in the contract work statement. Analytical studies established the control equations for these missions.

##### 2. Information Content Study

The information required for situation assessment and trajectory and attitude control for the mission phases under study was established for three levels of display comprehensiveness. Allocation of information to projection and panel display was made on the basis of specific criteria.

##### 3. Cockpit Mock-up

Full size simulated spacecraft cockpits were made for utilization in the Projection Display/Panel Display Integration Study and the Display/Control Compatibility Study (Appendix B).

##### 4. Projection Display/Panel Integration Study

Several examples of display systems with appropriate information content in the projection display and in the panel display were established. Information redundancy, display compatibility and failure detection were considered.

## 5. Display/Control Compatibility Study

The compatibility of the display configuration with anticipated flight control techniques was studied considering displacement, rate and acceleration type control systems.

## 6. Equipment Trade-Off Studies

Electronic equipment trade-off studies for three levels of display comprehensiveness were conducted. Size, weight, power and reliability were covered, as well as the effects of various sensor input types.

Optical system design and application charts were generated for use in projection display application studies. Volume and weight trade-off curves were made for basic system application parameters.

## 3-2. MISSION DESCRIPTION

The preliminary design investigation has been directed to two critical areas of manned space flight in which pilot capability, augmented with proper display information, may be used to increase mission success probability. These areas are the terminal phases of rendezvous and the final approach and letdown phases of lunar landing. The display concepts may also be used for other mission phases as well, although they are not specifically covered in the study.

### A. RENDEZVOUS

The rendezvous operation is taken as that phase of orbital flight during which a rendezvous vehicle and a target vehicle are within radar contact of each other, and the rendezvous vehicle is maneuvering to approach and dock with the target. The two major phases of the rendezvous mission are "terminal" and "docking". The terminal phase extends to a point within several hundred feet of the target, during which period instrumentation is required for control and assessment. The docking phase is primarily one of visual control and assessment. The terminal phase includes sightline and braking control.

There are several rendezvous techniques which are under consideratior for missions such as Gemini and Apollo, and these are described in Appendix C. In all cases the very final stage of the maneuver will in all probability be the same, i.e., visual line of sight rate and range rate control to completion of docking. The differences in the early phases of terminal rendezvous are in the type of control, i.e., proportional navigation (closed loop control) or orbital mechanics (open loop), and variations of these.

#### 1. Rendezvous Techniques

##### a. Proportional Navigation

In the proportional navigation rendezvous, the rotation rate of the relative velocity vector is continuously controlled as a function of line of sight (LOS) rate, and the range rate is decreased as a function of range. The vehicle acceleration ($a_c$) required to satisfy both conditions is defined by equation C-1 in Appendix C which specifies the magnitude and direction of the required thrust ($\bar{T} = a_c$ x vehicle mass). The pilot control requirement is the alignment of the thrust axis in the prescribed and continuously changing direction, while modulating the thrust magnitude according to the computed command. Fundamental information requirements are the command attitude, command thrust, LOS rates, range-to-go and range rate.

In general, the dynamics of proportional navigation control reduce the LOS rate to zero early in the maneuver after which control is primarily that to maintain zero LOS rate while following the braking thrust program. A simplified control scheme is, therefore, applicable with small increase in fuel

expenditure, in which the gross line of sight rate is first reduced to a small value without regard to range rate, after which range rate is controlled with necessary vernier LOS rate corrections. In vehicles with a main engine and vernier translational thrusters, high initial LOS rates may be beyond the capability of translational thrusters for control in reasonable time periods   In this case, the main engine thrust may be used for the LOS rate correction, after which the vehicle may be rotated to permit use of the same engine for range rate braking control. For control of this type of rendezvous, the initial thrust axis direction must be indicated to the pilot to permit thrusting in the proper direction in the plane at right angles to the LOS to cancel the LOS rate. Furthermore, the target direction must be unambiguously displayed for the rapid visual re-acquisition of the target for subsequent control.

As the rendezvous progresses and the pilot's assessment of the acceptability of the maneuver becomes highly dependent on his visual observations, a stabilized reference frame in his field of view, aligned with the target, is desirable to enhance his perception of LOS rate. The direction of the thrust axis in the visual field is required to insure that the braking thrust impulses do not cause unwanted LOS rates. Display of the thrust vector direction in the visual field would also be advantageous in that it may be easily tilted in a direction opposite to the relative target LOS motion while braking, to utilize the braking thrust for LOS control with small effect on the range rate.

b. Orbital Mechanics

The orbital mechanics technique is the classic two-impulse orbital transfer maneuver. The first impulse places the rendezvous vehicle on a collision course with the target vehicle, and the second impulse reduces the relative velocity to zero at a short range from the target. For a given time interval to rendezvous, this method is the most efficient from a fuel utilization standpoint; however, the required computer complexity and sensor accuracies are high. Since the operation is open-loop in the sense that any errors are not controlled on a continuous basis, the assessment display requirements are crucial. The pilot must have a basis for quantitatively monitoring the systems performance. Such quantities as the required and the actual velocity increment and the required and actual velocity direction, as well as time-to-go to the braking impulse, are important. The direction of the braking thrust (opposite the relative velocity vector), the velocity impulse required, and the actual closure range-rate must be clearly indicated to the pilot. Since he will rely on his own observations thru the window for assessment, display of critical information in the window is desirable.

### c. Modified Orbital Mechanics

Reference 8 describes an analysis of miss distance boundaries for 2-impulse rendezvous, in which the initial impulse is in the direction of the target. For this case, the miss distance ($\Delta R$) is a simple function of the orbital rotation rate ($\omega$) and time-to-go (t) to the rendezvous:

$$\frac{\Delta R}{R_O} = \omega t$$

in which $R_O$ is the initial range from target. This equation leads to the possibility of using an offset angle for the desired thrust equal to $\omega t$ (in radians, for small angles). If the offset is selected on the basis of the orbital rotation rate and a time-to-go (from the vehicle accelerations to be used), the miss distance should be minimized. The simplicity of the equation lends itself to a "flight handbook" technique in which the pilot can determine a thrusting direction offset based on his range, $R_O$, $\omega$, and time to rendezvous. For large ranges (and thus long time-to-go situations), the required angular offset may become large, and therefore the accuracy will be decreased. However, for $\omega t$ in the range of 5 to 10 degrees, the method should work well, while requiring a minimum of sensors and computation. A simple gunsight type display indicating angular offsets can be used for the thrust line offset in such a system.

### 2. Rendezvous Reference Systems

The reference system used to establish a collision course in a direct visual rendezvous will depend on the background against which the target vehicle is viewed. Conventionally, inertial reference systems are generally considered, using stars for the reference frame. A stable reference frame is required since the astronaut must establish the translational motion of his spacecraft relative to the target. This can only be done if the astronaut first stops his rotation with respect to a stable frame of reference which can be seen. It appears likely that only the horizon and the stellar field can meet this stability requirement. The earth below the spacecraft is moving angularly at a rate of approximately 170 degrees per minute for an orbital altitude of 100 miles. This angular rate precludes its use as a stable background. When looking up at the target against a star background, the astronaut can maneuver to maintain the target against this background. However, this operation may not be as efficient as the case where the target is maintained in a fixed position relative to the local vertical. To perform the latter operation, the visual field must be augmented with instrumentation to provide a stable reference.

### 3. Docking

The final rendezvous phase, docking, is that maneuver in which visual control of the rendezvous vehicle is used for vernier positional and attitude adjustments for actual contact. This phase is considered to cover the final several hundred feet of range. Line of sight rate control and range rate control are required to a point close to the target from which the final translations and rotations align the docking systems for mating. It is likely that no strong requirement exists for display information during this phase, with the exception of attitude, which will enhance the maneuver through stable control of thruster direction.

### B. LUNAR LANDING

Three phases of lunar landing will be treated, with emphasis on the latter two. These phases are:

- Initial descent from perilune at approximately 50,000 feet to approximately 1000 feet. This is the primary braking phase.

- Approach from approximately 1000 feet to the near vicinity of a touchdown area at under 100 feet in altitude.

- Letdown along a near vertical path to touchdown.

### 1. Initial Descent Phase

A transfer orbit places the lunar vehicle in a grazing path approximately 50,000 feet above the moon (lunar surface). From this point (perilune) the initial descent phase starts and continues to some low altitude on the order of 1000 feet. This phase is primarily a braking phase in which the speed is reduced from orbital speed to nearly zero to permit final visual survey of prospective landing areas.

This phase may be automatically controlled or accomplished manually by the astronaut. For a "gravity turn" type of descent, control requirement is to maintain continuous alignment of the braking thrust vector with the flight path, while a thrust program reduces the velocity in a predetermined manner to the desired value at approximately 1000 feet.

It is desirable during this phase that the vehicle be oriented to permit visual reference to the lunar surface as the altitude decreases, since this will be the first opportunity for significant visual observations. The significant attitude and thrust command information should be in the window during this period, whether automatic or manual flight control is involved, to permit continuous assessment of the progress of the descent while viewing the lunar surface. Quantitative velocity, altitude, and descent rate data must be available to the astronaut for continuous observation, although not necessarily in the window.

## 2. Approach Phase

The final approach phase from approximately 1000 feet altitude to under 100 feet will in all probability be along a relatively shallow (10 to 20 degrees) straight line approach path. There are several reasons why such a path is desirable. In reference 1, for example, the authors point out pilot preference for such an approach path for the following reasons:

- The constant angle path makes it easier to detect deviations from the desired conditions

- Fewer thrust and attitude control adjustments are required

- Rates are easier to judge

- The small declination of landing area from the horizon make flight path easy to maintain.

Further, the shallow path permits close-up viewing of the lunar surface during the approach, permitting a better evaluation of the probable landing area characteristics. The shallow path also reduces the field of view requirement for the window, because of the proximity of the landing area to the horizon.

The approach phase is initiated from a position with very low vertical and translational rates. The pilot uses this short period to make a final touchdown area selection and in all probability he will initiate a straight line descent to the selected area from his present position. A display of the direction of vehicle velocity is desirable in the astronaut's external visual field during this approach initiation interval. Alignment of the velocity vector with the touchdown area will insure a smooth transition to the descent and minimize fuel expenditure.

During the descent, stable control of flight path and speed (rate of descent) is necessary to avoid excessive fuel expenditure and to insure controlled end conditions near the ground. Since the astronaut will use this final approach phase for continuous and careful observation of the intended landing site, the inclusion of critical path and speed control information in the window, suitably integrated with the external scene, should provide marked improvement in control and risk assessment. The terrain will be unfamiliar to the pilot, and no previous "real" experience will be available to him to furnish a background from which displacement and rate information may be obtained with assurance. In addition, the time interval during which the final evaluation of the landing terrain must take place is short. It is, therefore, clearly desirable that the critical control and assessment information be furnished in the pilot's external field of view. As the vehicle approaches the

landing site, closer observation of this area may require modification of the flight path, and this should be easily accomplished through the displays furnished the pilot.

If the landing area is homogeneous, the straight line descent can terminate at a low altitude, say 10 to 20 feet, and a vertical letdown can be accomplished through rate of descent control only, without actual geographic touchdown position control during this short time interval. If close-up observation of the intended touchdown area dictates a change in location, a final horizontal flight phase may be used at low altitude prior to the final vertical descent.

Critical information during this descent phase, in addition to the thrust and path commands, are attitude, altitude, rate of descent, and the position and positional rates relative to the desired path. Altitude and rate of descent are primarily situation assessment quantities unless path and thrust command displays are not in use, in which case they become primary control data. Attitude is basic information necessary for control at all times.

Fundamental to the use of projected displays for lunar landing are the basic flight path and speed control aspects. Previous programs conducted by Sperry have shown the benefits attainable in stable final approach flight control through the use of a path and speed control display which is utilized in conjunction with the real world. These concepts have been proven in military fighter aircraft (references 2 and 3). These vehicles operate in an aerodynamic flight regime, lunar landing vehicles do not. Therefore, a preliminary analog computer simulation was undertaken to establish the feasibility of the path and speed control concepts for a lunar landing vehicle. A description of this study and its results are given in Appendix C. In summary, the use of a minimum number of sensors (attitude, vertical acceleration and altitude) as inputs to computers which were synthesized has resulted in stable, easily controlled approach trajectories for a LEM-type vehicle. Control was attained by the tracking of an aiming point on the lunar surface with a projected image. The combination of this path control image and a secondary speed control element furnished both pitch attitude and thrust command information to the pilot during the approaches. The actual display techniques used are further described in subparagraph 3-3 and Appendix C.

3. Letdown Phase

The final landing phase is a vertical descent to touchdown. The vertical descent is necessary to minimize translational velocity. The maneuver consists of a controlled vertical rate as a function of altitude with translational rates kept under 5 fps. Vertical rate is expected to be under 10 fps. Precise pitch and roll attitude control is required to minimize translational accelerations.

Tight control of vertical rate is necessary to insure a touchdown within the structural strength envelope of the spacecraft. The actual touchdown attitude must be controlled within limits. Primary display requirements are attitude, vertical and translational rates, altitude and probably a letdown director for stable vertical rate control. In a normal visual landing, translational and vertical rates can be adequately judged near the ground through reference to the visual scene. In the lunar landing case dust clouds may obscure the real world view making imperative effective instrument displays in the window which may be used without a display transition at this critical point.

## 3-3. INFORMATION CONTENT

### A. GENERAL

The information content study has established the basic information requirements for situation assessment and trajectory and attitude control for the mission phases under study. Specifically, these are the initial descent, the approach and the letdown modes of lunar landing, and the initial (LOS rate arrestment) and the braking modes of rendezvous.

The study established:

(1)    The information to be considered

(2)    The information classification to be used in the analysis

(3)    Criteria for the assignment of the information to the projected display

(4)    Criteria on the form of any information which is used in the projected display

(5)    Criteria for assignment of information to the panel display.

Operations on the information with these classifications and criteria were completed and recommended display configurations for each mode were generated.

### B. DISPLAY SYSTEM CONFIGURATIONS CONSIDERED

Three display system configurations, each with increasing levels of display comprehensiveness, were considered in the study.

#### 1.  First Level (Display System 1)

Simple flight path control display which includes an additional symbolic element. The path control element is positioned against the usually perceived background, such as the rendezvous target, for control. This is a display with two moving symbolic elements. The first symbol has two degrees of positional freedom, i.e., vertically and laterally, in the field of view and has no rotational freedom. The second symbol has only one degreee of freedom (vertical) relative to a fixed reference point or to the first symbolic element.

2. Second Level (Display System 2)

The second level is a dual channel and dual color display system which generates a control and situation display. Situation elements are shown as simple cues such as dots, circles, lines, etc. This is a display which consists of:

   a. First level (Display System 1) reticle generated information of one color, plus

   b. CRT generated information of a second color. The CRT generated cues have three degrees of positional freedom i. e., vertical, lateral and rotational freedom.

3. Third Level (Display System 3)

This display is the same as Display System 2 except that situation elements will show true size, shape and perspective.

Director (or quickened) types of control information will be considered in each case.

C. SPACECRAFT TYPES CONSIDERED

In reviewing display-control techniques for spacecraft applications, the physical orientation of the pilot's normal field of view, relative to the direction of the thrust vector of the engine, must be considered.

For this program, a spacecraft similar to the Gemini or Apollo command module types was selected for the rendezvous mission study, i. e., a spacecraft with:

   (1) An acceleration (main) engine and a braking engine with thrust lines 180 degrees apart

   (2) The pilot's normal field of view approximately parallel to and in the direction of the main engine thrust vector

A spacecraft similar to the Apollo LEM type was selected for the lunar landing study, i. e., a spacecraft with:

   (1) A main engine that may be used for vertical, lateral and positional displacement control near the lunar surface through use of vehicle attitude and thrust control

   (2) The pilot's normal field of view is approximately 90 degrees from the main engine thrust direction.

Application of the projected display technique to telescopic or periscope devices in which wide fields of view or fields of view with a direction other than in line with the direction of the pilot's normal field of view is not specifically covered in this report, although the techniques are applicable to such systems.

## D. INFORMATION ANALYSIS

A study was conducted of the functions which the pilot must perform during the course of the spacecraft mission. The information required to perform each of these functions was determined, and an analysis of the operations required on this information was conducted. Criteria for the assignment of information to the projected display and the cockpit panel display were established; these are described below.

Table 3-3.1 shows the flight control and assessment information considered in this study. Other required information utilized by the pilot in the spacecraft (information on fuel subsystem, propulsion subsystem, environmental subsystem) has not been included since it is beyond the scope of this contract. The study is specifically confined to the primary displays and controls for satisfactory flight operations by the pilot during the mission.

To analyze the information, the following categories were established:

- Mission phase - rendezvous or lunar landing

- Task - (A) attitude, (N) navigation, (P) propulsion or (A) environmental or auxiliary system task

- Purpose - (A) assessment or (C) control

- Frequency of use - (1) high, (2) intermediate, (3) low sampling rate

- Type - qualitative (QUAL) or quantitative (QUAN)

- Importance - (N) necessary, (D) desirable, or (U) unnecessary

The information was classified into the above categories. Tables 3-3.2 and 3-3.3 are a summary of the study of the information for the rendezvous mission and tables 3-3.4 through 3-3.6 cover the lunar landing mission.

It should be recognized that information charts are, in reality, a check list and that the final display must necessarily include other factors which may not be readily adapted to classification in tabular form.

## E. ALLOCATION OF INFORMATION TO THE PROJECTED DISPLAY AND COCKPIT PANEL DISPLAY

### 1. Criteria for Allocation of Information to the Projected Display

Heads-up flight in which the pilot extracts important information from the real world has a richness in content and a high degree of organization for perceptual assimilation. The pilot has had lifelong experience in handling information in this form.

TABLE 3-3.1

INFORMATION CONSIDERED

| Displacement | Rates | Accelerations | Error and Commands |
|---|---|---|---|
| Roll | Roll rate | Roll acceleration | Roll error |
| Pitch | Pitch rate | Pitch acceleration | Pitch error |
| Yaw | Yaw rate | Yaw acceleration | Yaw error |
| Flight Path (FPA) (Vertical and Lateral Total Velocity) | FPA rate | - | FPA error |
| Target Elevation | Target elevation rate | - | Elevation error |
| Target Azimuth | Target azimuth rate | - | Azimuth error |
| Target range | Target range rate | - | |
| X Displacement | X velocity | X acceleration | |
| Y Displacement | Y velocity | Y acceleration | |
| Z Displacement | Z velocity | Z acceleration | |
| Main Thrust | - | - | Thrust error |
| Fuel Available | Fuel rate | | |
| Orbital Time | - | - | Altitude Director |
| Event Elapsed Time | - | - | Thrust Director |
| Greenwich Time | - | - | Path Control Director |
| | | | Speed Control Director |
| | | | Vertical Velocity Director |

Precise criteria for the assignment of information to the projected display can be established on the basis of the foregoing. When heads-up flight is considered advantageous, what information should be included in the projected display? The criteria which are considered to be the most significant were classified and identified as below.

Tables 3-3.2 through 3-3.6 summarize the information included in the display and the specific criteria, as defined in the following paragraph, for inclusion of the information in the projected display.

These criteria are defined as follows:

- P - Information which enhances the precision of visual flight control.

  Example: Providing projected display information which augments the pilot's ability to control the flight path of the vehicle in relation to the aiming point on the lunar landing area.

- A - Information which will improve the ability of the pilot to assess an existing situation when information from the external visual field is available.

  Example: (1) The display of the spacecraft velocity vector direction in the projected display will enable the pilot to observe the relationships of the flight path of the vehicle to the lunar surface.

  (2) A stabilized reference cue which furnishes an index to determine target motion for rendezvous.

- I - Information which will enhance "instrument" flight, through either a more favorable perceptual arrangement of displayed information in a three-dimensional context, or where transition from instrument to visual flight must be accomplished rapidly and smoothly.

  Example: (1) Tracking a radar target defining a rendezvous vehicle may be more easily accomplished with the target and a quickened flight control response element displayed in the projected display. Visual acquisition of the target may then be accomplished smoothly and a transition to visual flight control effected with continuity in the display/control procedure.

  (2) Providing an aiming point cue for lunar landing. A smooth transition to instrument flight control may then be affected if lunar dust obscures the "real world" guidance cues prior to touchdown.

● S - Information which must be sampled frequently on a real-time basis during heads-up flight control.

Example: (1) Ground speed must be controlled closely during approach to the hover condition. If tight flight path control is being achieved with a projected display, it is reasonable to include speed control information in the same display. Frequent shifts in visual attention between the panel and the projected display will thereby be avoided.

(2) Sight line rate, range rate and attitude control in rendezvous maneuvering during terminal phases.

The frequency criterion should be distinguished clearly from the matter of importance of the information. Information, which has appreciable time rates of change and a variation in mean value, must be sampled frequently if the variable is to be controlled. Continuous control of an error in a tracking task represents the extreme case for this type of information. However, there are variables such as suit temperature which are quasistatic though extremely important to the success of the mission. These need not be presented in the projected display, but suitable warning, either audible, a lamp indication on the panel, or a visual master warning alarm in the projected display, is required to indicate when the mission accomplishment is in jeopardy.

2. Criteria on the Form or Method of Displaying the Information in the Projected Display

These are:

● Display elements which have their visual counterparts in the real world must be in registery with same.

Example: A symbolic image of the landing area must be superimposed on the real landing area viewed through the window.

● Scale factors for the moving elements related to the real world are fixed to yield a 1:1 dynamic relationship.

Example: The motion of a simulated horizon accompanying changes in pitch must be identical to the perceived motion of the real horizon.

● The number of display elements should be minimized, and their configurations should be made simple to reduce any occlusion of the visual field.

- A specific cue form should be used for one function only in all phases of flight.

- Special consideration must be given to the uniqueness of spacecraft operation as compared to conventional aircraft operation. For the spacecraft operation, the vehicle may be oriented in any direction relative to the flight path and the control about the various axes is independent. Therefore, special emphasis must be placed on employing favorable perceptual arrangements of displayed information in a three dimensional context.

3. Criteria for Assignment of Information to the Panel Display

These are:

- The panel display and the real world view are the primary flight assessment and control displays with a non-operating projected display.

- The panel display alone may be the primary flight assessment and control display with a non-operating projected display and poor real world visibility.

  Example: A non-operating projected display and dust clouds during the final lunar letdown.

- The panel displays:

  (1) are backup displays for flight assessment and flight control when the projected display is in use

  (2) are the primary displays for vehicle systems conditions at all times

  (3) provide a medium for display of situation and flight control information which is less demanding in terms of visual reference to the real world or in required sampling rate

  (4) provide a medium for detailed quantitative information on total spacecraft flight situation, flight control and spacecraft systems condition

  (5) provide a medium for display of information of the real world, which cannot be on a 1:1 scale with the real world, due to the large angles involved and the necessary practical limits on the spacecraft window field of view.

The material presented below defines the nomenclature used in tables 3-3.2 through 3-3.6.

Code on classification of information:

<u>Type</u>: (QUAL) qualitative, (QUAN) quantitative

<u>Task</u>: (A) attitude, (N) navigation, (P) propulsion, (E) environmental or auxiliary system

<u>Purpose</u>: (A) assessment, (C) control

<u>Frequency of Use</u>: (1) high, (2) intermediate (3) low sampling rate

<u>Importance</u>: (N) necessary, (D) desirable, (U) unnecessary

<u>Applicability of Info</u>: (X) indicates information is used. (-) indicates information is not used.

<u>Criteria for Use in Projected Display</u>:
(P) enhancement of the precision of visual flight control
(A) improvement of assessment ability when external visual field information is available
(S) required frequent sampling on a real time basis during heads up-flight control
(I) enhancement of "instrument" flight

## F. RECOMMENDED DISPLAY CONFIGURATIONS

Typical mission phases, spacecraft flight characteristics and the dynamics and control techniques required for each of the mission modes have been described in paragraph 3-2 and appendix C. A projected display philosophy has been described and an information analysis conducted.

Specific recommendations for displays which implement the information study, the cue forms used to make the displays, and their operation are described in the following sections.

To aid in the discussion and understanding of the recommended display configurations, the basic objective and control operation required for each mode is repeated in simplified form.

Many types of possible flight profiles and control exist (i.e., all automatic, automatic and manual, all manual). It would not be possible, within the work scope of the present contract, to present the operational procedures for the use of projected displays for the many possible cases. Therefore, displays and controls, and their operation for selected typical cases, are presented to illustrate the applicability of the projected display

TABLE 3-3.2 INFORMATION ANALYSIS SUMMARY

## RENDEZVOUS — INITIAL LOS RATE ARRESTMENT MODE

| INFO | TYPE | TASK (A,N P,E) | PURPOSE A,C | FREQ OF USE 1 2 3 | IMPOR-TANCE N,D,U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P A S I |
|------|------|------|------|------|------|------|------|
| ROLL | QUAL | A | A | 3 | D | X' | — |
|  | QUAN | A | A | 3 | D | X | — |
| PITCH | QUAL | A | A | 3 | D | X | — |
|  | QUAN | A | A | 3 | D | X | — |
| YAW | QUAL | A | A | 3 | D | X | — |
|  | QUAN | A | A | 3 | D | X | — |
|  |  |  |  |  |  |  |  |
| FPA | QUAL. | — | — | — | U | — | — |
|  | QUAN | — | — | — | U | — | — |
| TOTAL VELOCITY | QUAL | — | — | — | — | — | — |
|  | QUAN | N | A | 2 | D | X | — |
|  |  |  |  |  |  |  |  |
| TARGET ELEV | QUAL | A N | A C | 2 | N | X | — |
|  | QUAN | A N | A C | 3 | N | X | ASI |
| TARGET AZIMUTH | QUAL | A N | A C | 2 | N | X | — |
|  | QUAN | A N | A C | 3 | N | X | ASI |
| TARGET RANGE | QUAL | N | A | 3 | N | X | — |
|  | QUAN | — | — | — | — | — | — |
|  |  |  |  |  |  |  |  |
| X DISPMNT | QUAL | — | — | — | U | — | — |
|  | QUAN | — | — | — | U | — | — |
| Y DISPMNT | QUAL | — | — | — | U | — | — |
|  | QUAN | — | — | — | U | — | — |
| Z DISPMNT | QUAL | — | — | — | U | — | — |
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| MAIN THRUST | QUAL | N P | A C | 1 | N | X | AIS |
|  | QUAN | N P | A | 2 | N | X | — |
| FUEL AVAIL | QUAL | ANPE | A C | 1 | N | X | — |
|  | QUAN | ANPE | A | 2 | N | Y | — |
|  |  |  |  |  |  |  |  |
| ORBITAL TIME | QUAL | — | — | — | U | — | — |
|  | QUAN | N E | A | 3 | D | X | — |
| EVENT ELAPSED TIME | QUAL | N P | A C | 2 | N | X | — |
|  | QUAN | N P | A C | 2 | N | X | — |
| GREENWICH TIME | QUAL | N E | A | 3 | D | — | — |
|  | QUAN | N E | A | 3 | D | X | — |
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| INFO | TYPE | TASK ANPE | PURPOSE A, C | FREQ OF USE 1 2 3 | IMPOR- TANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P, A, S, I | |
|---|---|---|---|---|---|---|---|---|
| ROLL RATE | QUAL | — | — | — | — | — | — | |
| | QUAN | A | A C | 2 | N | X | — | |
| PITCH RATE | QUAL | — | — | — | — | — | — | |
| | QUAN | A | A C | 2 | N | X | — | |
| YAW RATE | QUAL | — | — | — | — | — | — | |
| | QUAN | A | A C | 2 | N | X | — | |
| | | | | | | | | |
| FPA RATE | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
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| | | | | | | | | |
| TGT ELEV RATE | QUAL | — | — | — | — | — | — | |
| | QUAN | A N | A C | 1 | N | X | A S I | |
| TGT AZ RATE | QUAL | — | — | — | — | — | — | |
| | QUAN | A N | A C | 1 | N | X | A S I | |
| TGT RGE RATE | QUAL | A N | A C | 2 | N | X | — | |
| | QUAN | N | A | 2 | N | X | — | |
| | | | | | | | | |
| X VELOCITY | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
| Y VELOCITY | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
| Z VELOCITY | QUAL | — | — | — | U | — | — | |
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| FUEL RATE | QUAL | A N P | A C | 1 | N | X | — | |
| | QUAN | A N P | A C | 2 | N | X | — | |

| INFO | TYPE | TASK A N P E | PURPOSE A,C | FREQ OF USE 1 2 3 | IMPOR-TANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA PASS | |
|------|------|------|------|------|------|------|------|------|
| ROLL | QUAL | A | A | 3 | D | X | — | |
| ACCEL | QUAN | — | — | — | — | — | — | |
| PITCH | QUAL | A | A | 3 | D | X | — | |
| ACCEL | QUAN | — | — | — | — | — | — | |
| YAW | QUAL | A | A | 3 | D | X | — | |
| ACCEL | QUAN | — | — | — | — | — | — | |
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| ACCE- | QUAN | — | — | — | — | — | — | |
| VEH Y | QUAL | E | A | 3 | D | X | — | |
| ACCEL | QUAN | — | — | — | — | — | — | |
| VEH Z | QUAL | E | A | 3 | D | X | — | |
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| INFO | TYPE | TASK A,N,PE | PURPOSE A,C | FREQ OF USE 1 2 3 | IMPORTANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P,A,S,I | |
|---|---|---|---|---|---|---|---|---|
| ROLL ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | A | A C | 2 | D | X | PSI | |
| PITCH ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | A N P | A C | 1 | N | X | PSI | |
| YAW ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | A N P | A C | 1 | N | X | PSI | |
| | | | | | | | | |
| FPA ERROR | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
| | | | | | | | | |
| | | | | | | | | |
| ELEV ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | A N | A C. | 2 | D | X | A I | |
| AZIMUTH ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | A N | A C | 2 | D | X | A I | |
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| THRUST ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | N P | A C | 1 | N | X | — | |
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| ATTITUDE DIR | QUAN | A N | C | 1 | N | X | PSI | |
| THRUST DIR | QUAN | N P | C | 1 | N | X | PSI | |
| PATH DIR | — | — | — | — | — | — | — | |
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## TABLE 3-3.3 INFORMATION ANALYSIS SUMMARY

### RENDEZVOUS - BRAKING MODE

| INFO | TYPE | TASK (A,N,P,E) | PURPOSE A,C | FREQ OF USE 1 2 3 | IMPOR-TANCE N,D,U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P A S I | |
|---|---|---|---|---|---|---|---|---|
| ROLL | QUAL | A | A | 3 | D | X | — | |
| | QUAN | A | A | 3 | D | X | — | |
| PITCH | QUAL | A | A | 3 | D | X | — | |
| | QUAN | A | A | 3 | D | X | — | |
| YAW | QUAL | A | A | 3 | D | X | — | |
| | QUAN | A | A | 3 | D | X | • | |
| | | | | | | | | |
| FPA | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
| TOTAL VELOCITY | QUAL | — | — | — | — | — | — | |
| | QUAN | N | A | 2 | D | X | — | |
| | | | | | | | | |
| TARGET ELEV | QUAL | A N | A C | 2 | N | X | — | |
| | QUAN | A N | A C | 3 | N | X | A S I | |
| TARGET AZIMUTH | QUAL | A N | A C | 2 | N | X | — | |
| | QUAN | A N | A C | 3 | N | X | A S I | |
| TARGET RANGE | QUAL | A N | A C | 1 | N | X | — | |
| | QUAN | A N | A C | 1 | N | X | A S I | |
| | | | | | | | | |
| X DISPMNT | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
| Y DISPMNT | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
| Z DISPMNT | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
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| MAIN THRUST | QUAL | N P | A C | 1 | N | X | A I S | |
| | QUAN | N P | A | 2 | N | X | — | |
| FUEL AVAIL | QUAL | A N P E | A C | 1 | N | X | — | |
| | QUAN | A N P E | A | 2 | N | X | — | |
| | | | | | | | | |
| ORBITAL TIME | QUAL | — | — | — | U | — | — | |
| | QUAN | N E | A | 3 | D | X | — | |
| EVENT ELAPSED TIME | QUAL | N P | A C | 2 | N | X | — | |
| | QUAN | N P | A C | 2 | N | X | — | |
| GREENWICH TIME | QUAL | N E | A | 3 | D | X | — | |
| | QUAN | N E | A | 3 | D | X | — | |
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| INFO | TYPE | TASK ANPE | PURPOSE A, C | FREQ. OF USE 1 2 3 | IMPOR- TANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P, A, S, I | |
|------|------|-----------|--------------|--------------------|--------------------|---------------|----------------------------------------|---|
| ROLL RATE | QUAL | — | — | — | — | — | — | |
| | QUAN | A | A C | 2 | N | X | — | |
| PITCH RATE | QUAL | — | — | — | — | — | — | |
| | QUAN | A | A C | 2 | N | X | — | |
| YAW RATE | QUAL | — | — | — | — | — | — | |
| | QUAN | A | A C | 2 | N | X | — | |
| | | | | | | | | |
| FPA RATE | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
| | | | | | | | | |
| | | | | | | | | |
| TGT ELEV RATE | QUAL | — | — | — | — | — | — | |
| | QUAN | A N | A C | 1 | N | X | A S I | |
| TGT AZ RATE | QUAL | — | — | — | — | — | — | |
| | QUAN | A N | A C | 1 | N | X | A S I | |
| TGT RGE RATE | QUAL | A N | A C | 1 | N | X | — | |
| | QUAN | N | A | 1 | N | X | A S I | |
| | | | | | | | | |
| X VELOCITY | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
| Y VELOCITY | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
| Z VELOCITY | QUAL | — | — | — | U | — | — | |
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| FUEL RATE | QUAL | A N P | A C | 1 | N | X | — | |
| | QUAN | A N P | A C | 2 | N | X | — | |

32

| INFO | TYPE | TASK A.N.P.E | PURPOSE A,C | FREQ OF USE 1 2 3 | IMPOR-TANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA PASI | |
|------|------|------|------|------|------|------|------|---|
| ROLL ACCEL | QUAL | A | A | 3 | D | X | — | |
| | QUAN | — | — | — | — | — | — | |
| PITCH ACCEL | QUAL | A | A | 3 | D | X | — | |
| | QUAN | — | — | — | — | — | — | |
| YAW ACCEL | QUAL | A | A | 3 | D | X | — | |
| | QUAN | — | — | — | — | — | — | |
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| VEH X ACCEL | QUAL | E | A | 3 | D | X | — | |
| | QUAN | — | — | — | — | — | — | |
| VEH Y ACCEL | QUAL | E | A | 3 | D | X | — | |
| | QUAN | — | — | — | — | — | — | |
| VEH Z ACCEL | QUAL | E | A | 3 | D | X | — | |
| | QUAN | | | | | | | |

| INFO | TYPE | TASK A,N,PE | PURPOSE A,C | FREQ PHASE 1 2 3 | IMPOR-TANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P A S I | |
|------|------|------|------|------|------|------|------|------|
| ROLL ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | A | AC | 2 | D | X | PSI | |
| PITCH ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | ANP | A C | 1 | N | X | PSI | |
| YAW ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | ANP | A C | 1 | N | X | PSI | |
| | | | | | | | | |
| FPA ERROR | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
| | | | | | | | | |
| | | | | | | | | |
| ELEV ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | A·N | A C | 2 | D | X | A I | |
| AZIMUTH ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | A N | A C | 2 | D | X | A I | |
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| THRUST ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | N P | A C | 2 | D | X | — | |
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| ATTITUDE DIR | QUAN | AN | C | 1 | N | X | PSI | |
| THRUST DIR | QUAN | N P | C | 1 | N | X | PSI | |
| PATH DIR | — | — | — | — | — | — | — | |
| SPEED DIR | — | — | — | — | — | — | — | |
| VER VEL DIR | — | — | — | — | — | — | — | |

# TABLE 3-3.4 INFORMATION ANALYSIS SUMMARY

## 2240 LUNAR LANDING – INITIAL DESCENT MODE

| INFO | TYPE | TASK (A,N,P,E) | PURPOSE A,C | FREQ OF USE 1 2 3 | IMPOR-TANCE N D,U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P,A,S,I |
|------|------|------|------|------|------|------|------|
| ROLL | QUAL | A | A | 2 | N | X | PAI |
|  | QUAN | A | A | 2 | N | X | — |
| PITCH | QUAL | A | A | 2 | N, | X | PAI |
|  | QUAN | A | A | 2 | N | X | — |
| YAW | QUAL | A | A | 2 | N | X | PAI |
|  | QUAN | A | A | 2 | N | X | — |
|  |  |  |  |  |  |  |  |
| FPA | QUAL. | ANP | AC | 2 | N | X | PAI |
|  | QUAN | ANP | AC | 2 | N | X | — |
| TOTAL VELOCITY | QUAL | — | — | — | — | — | — |
|  | QUAN | N P | AC | 2 | N | X | — |
|  |  |  |  |  |  |  |  |
| TARGET ELEV | QUAL | — | — | — | — | — | — |
|  | QUAN | — | — | — | — | — | — |
| TARGET AZIMUTH | QUAL | — | — | — | — | — | — |
|  | QUAN | — | — | — | — | — | — |
| TARGET RANGE | QUAL | — | — | — | — | — | — |
|  | QUAN | — | — | — | — | — | — |
|  |  |  |  |  |  |  |  |
| X DISPMNT | QUAL | — | — | — | U | — | — |
|  | QUAN | N | A | 2 | N | X | — |
| Y DISPMNT | QUAL | — | — | — | U | — | — |
|  | QUAN | N | A | 2 | N | X | — |
| Z DISPMNT | QUAL | — | — | — | U | — | — |
|  | QUAN | N | A | 2 | N | X | — |
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| MAIN THRUST | QUAL | N P | A C | 1 | N | X | AIS |
|  | QUAN | N P | A | 2 | N | X | — |
| FUEL AVAIL | QUAL | ANPE | A C | 1 | N | X | — |
|  | QUAN | ANPE | A | 2 | N | X | — |
|  |  |  |  |  |  |  |  |
| ORBITAL TIME | QUAL | — | — | — | — | — | — |
|  | QUAN | N E | A | 3 | D | X | — |
| EVENT ELAPSED TIME | QUAL | N P | A C | 2 | N | X | S |
|  | QUAN | N P | A C | 2 | N | X | — |
| GREENWICH TIME | QUAL | N E | A | 3 | D | X | — |
|  | QUAN | N E | A | 3 | D | X | — |
|  |  |  |  |  |  |  |  |

35

| INFO | TYPE | TASK ANPE | PURPOSE A, C | FREQ CRUSE 1 2 3 | IMPORTANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P, A, S, I |
|---|---|---|---|---|---|---|---|
| ROLL RATE | QUAL | A  N | A C | 2 | D | X | — |
|  | QUAN | A  N | A C | 2 | N | X | — |
| PITCH RATE | QUAL | A  N | A C | 2 | D | X | — |
|  | QUAN | A  N | A C | 2 | N | X | — |
| YAW RATE | QUAL | A  N | A C | 2 | D | X | — |
|  | QUAN | A  N | A C | 2 | N | X | — |
|  |  |  |  |  |  |  |  |
| FPA RATE | QUAL | — | — | — | U | — | — |
|  | QUAN | — | — | — | U | — | — |
|  |  |  |  |  |  |  |  |
|  |  |  |  |  |  |  |  |
| TGT ELEV RATE | QUAL | — | — | — | — | — | — |
|  | QUAN | — | — | — | — | — | — |
| TGT AZ RATE | QUAL | — | — | — | — | — | — |
|  | QUAN | — | — | — | — | — | — |
| TGT RGE RATE | QUAL | — | — | — | — | — | — |
|  | QUAN | — | — | — | — | — | — |
|  |  |  |  |  |  |  |  |
| X VELOCITY | QUAL | — | — | — | — | — | — |
|  | QUAN | N | A | 2 | D | X | — |
| Y VELOCITY | QUAL | — | — | — | — | — | — |
|  | QUAN | N | A | 2 | D | X | — |
| Z VELOCITY | QUAL | — | — | — | — | — | — |
|  | QUAN | N | A | 2 | D | X | — |
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| FUEL RATE | QUAL | A N P | A | 1 | N | X | — |
|  | QUAN | A N P | A | 2 | N | X | — |

| INFO | TYPE | TASK A.N.P.E | PURPOSE A.C. | FREQ CRUSE 1 2 3 | IMPOR-TANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P A,S,I | |
|---|---|---|---|---|---|---|---|---|
| ROLL | QUAL | A | A | 3 | D | X | — | |
| ACCEL | QUAN | — | — | — | — | — | — | |
| PITCH | QUAL | A | A | 3 | D | X | — | |
| ACCEL | QUAN | — | — | — | — | — | — | |
| YAW | QUAL | A | A | 3 | D | X | — | |
| ACCEL | QUAN | — | — | — | — | — | — | |
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| VEH X | QUAL | N E | A | 3 | D | X | — | |
| ACCEL | QUAN | — | — | — | U | — | — | |
| VEH Y | QUAL | N E | A | 3 | D | X | — | |
| ACCEL | QUAN | — | — | — | U | — | — | |
| VEH Z | QUAL | N E | A | 3 | D | X | — | |
| ACCEL | QUAN | — | — | — | U | — | — | |
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| INFO | TYPE | TASK A,N,P,E | PURPOSE A,C | FREQ OF USE 1 2 3 | IMPOR-TANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA PA S,I | |
|---|---|---|---|---|---|---|---|---|
| ROLL ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | A N P | A C | 1 | N | X | A I | |
| PITCH ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | A N P | A C | 1 | N | X | A I | |
| YAW ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | A N P | A C | 1 | N | X | A I | |
| | | | | | | | | |
| FPA ERROR | QUAL | — | — | — | — | — | — | |
| | QUAN | — | — | — | N | X | — | |
| | | | | | | | | |
| | | | | | | | | |
| ELEV ERROR | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
| AZIMUTH ERROR | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
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| THRUST ERROR | QUAL | L | — | — | — | — | — | |
| | QUAN | N P | A | 3 | D | X | — | |
| | | | | | | | | |
| ATTITUDE DIR | QUAN | A N P | C | 1 | N | X | PS I | |
| THRUST DIR | QUAN | A N P | C | 1 | N | X | PS I | |
| PATH DIR | — | — | — . | — | — | — | — | |
| SPEED DIR | — | — | — | — | — | — | — | |
| VER. VEL. DIR | — | — | — | — | — | — | — | |

# TABLE 3-3.5 INFORMATION ANALYSIS SUMMARY
## LUNAR LANDING — APPROACH MODE

22.10

| INFO | TYPE | TASK (A,N,P,E) | PURPOSE A,C | FREQ OF USE 1 2 3 | IMPORTANCE N,D,U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P,A,S,I |
|---|---|---|---|---|---|---|---|
| ROLL | QUAL | A | A | 2 | N | X | A |
|  | QUAN | A | A | 2 | N | X | — |
| PITCH | QUAL | A | A | 2 | N | X | A |
|  | QUAN | A | A | 2 | N | X | — |
| YAW | QUAL | A | A | 2 | N | X | A |
|  | QUAN | A | A | 2 | N | X | — |
|  |  |  |  |  |  |  |  |
| FPA | QUAL | N | A | 2 | N | X | — |
|  | QUAN | N | A | 2 | N | X | — |
| TOTAL VELOCITY | QUAL | N | A | 2 | N | X | — |
|  | QUAN | N | A | 2 | N | X | — |
|  |  |  |  |  |  |  |  |
| TARGET ELEV | QUAL | — | — | — | — | — | — |
|  | QUAN | N P | C | 1 | N | X | P I |
| TARGET AZIMUTH | QUAL | — | — | — | — | — | — |
|  | QUAN | N P | C | 1 | N | X | P I |
| TARGET RANGE | QUAL | — | — | — | — | — | — |
|  | QUAN | N | A | 3 | D | X | — |
|  |  |  |  |  |  |  |  |
| X DISPMNT | QUAL | N P | A | 2 | N | — | A I |
|  | QUAN | N P | A | 3 | N | X | — |
| Y DISPMNT | QUAL | N P | A | 2 | N | — | A I |
|  | QUAN | N P | A | 3 | N | X | — |
| Z DISPMNT | QUAL | N P | A | 2 | U | — | A I |
|  | QUAN | N P | A | 3 | N | X | — |
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| MAIN THRUST | QUAL | NP | A | 2 | N | X | — |
|  | QUAN | NP | A | 2 | N | X | — |
| FUEL AVAIL | QUAL | ANPE | A | 1 | N | X | S |
|  | QUAN | ANPE | A | 1 | N | X | — |
|  |  |  |  |  |  |  |  |
| ORBITAL TIME | QUAL | — | — | — | U | — | — |
|  | QUAN | — | — | — | U | — | — |
| EVENT ELAPSED TIME | QUAL | N P | A | 2 | N | X | — |
|  | QUAN | N P | A | 2 | N | X | S |
| GREENWICH TIME | QUAL | — | — | — | U | — | — |
|  | QUAN | — | — | — | U | — | — |

39

| INFO | TYPE | TASK ANPE | PURPOSE A, C | FREQ CHNGE 1 2 3 | IMPOR-TANCE N D L | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P, A, S, I | |
|------|------|-----------|--------------|-------------------|--------------------|---------------|---------------------------------------|---|
| ROLL RATE | QUAL | A | A | 3 | D | X | — | |
| | QUAN | A | A | 3 | N | X | — | |
| PITCH RATE | QUAL | A | A | 3 | D | X | — | |
| | QUAN | A | A | 3 | N | X | — | |
| YAW RATE | QUAL | A | A | 3 | D | X | — | |
| | QUAN | A | A | 3 | N | X | — | |
| | | | | | | | | |
| FPA RATE | QUAL | — | — | — | L | — | — | |
| | QUAN | — | — | — | L | — | — | |
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| TGT ELEV RATE | QUAL | — | — | — | L | — | — | |
| | QUAN | — | — | — | L | — | — | |
| TGT AZ RATE | QUAL | — | — | — | L | — | — | |
| | QUAN | — | — | — | L | — | — | |
| TGT RGE RATE | QUAL | — | — | — | L | — | — | |
| | QUAN | — | — | — | L | — | — | |
| | | | | | | | | |
| X VELOCITY | QUAL | — | — | — | — | — | — | |
| | QUAN | N | A | 2 | N | X | — | |
| Y VELOCITY | QUAL | — | — | — | — | — | — | |
| | QUAN | N | A | 2 | N | X | — | |
| Z VELOCITY | QUAL | — | — | — | — | — | — | |
| | QUAN | N E | A | 2 | N | X | — | |
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| FUEL RATE | QUAL | A N P | A | 1 | N | X | — | |
| | QUAN | A N P | A | 2 | N | X | — | |
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| INFO | TYPE | TASK A,N,P,E | PURPOSE A,C | FREQ CRUISE 1 2 3 | IMPORTANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA PASS | |
|---|---|---|---|---|---|---|---|---|
| ROLL ACCEL | QUAL | A | A | 3 | D | X | — | |
| | QUAN | — | — | — | — | — | — | |
| PITCH ACCEL | QUAL | A | A | 3 | D | X | — | |
| | QUAN | — | — | — | — | — | — | |
| YAW ACCEL | QUAL | A | A | 3 | D | X | — | |
| | QUAN | — | — | — | — | — | — | |
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| VEH X ACCEL | QUAL | — | — | — | U | — | — | |
| | GUAN | E | A | 3 | D | X | — | |
| VEH Y ACCEL | QUAL | — | — | — | U | — | — | |
| | QUAN | E | A | 3 | D | X | — | |
| VEH Z ACCEL | QUAL | — | — | — | U | — | — | |
| | GUAN | E | A | 3 | D | X | — | |
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| INFO | TYPE | TASK A,N,PE | PURPOSE A,C | FREQ OF USE 1 2 3 | IMPORTANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P A & I |
|---|---|---|---|---|---|---|---|
| ROLL ERROR | QUAL | — | — | — | U | — | — |
| | QUAN | — | — | — | U | — | — |
| PITCH ERROR | QUAL | — | — | — | U | — | — |
| | QUAN | — | — | — | U | — | — |
| YAW ERROR | QUAL | — | — | — | H | — | — |
| | QUAN | — | — | — | U | — | — |
| | | | | | | | |
| FPA ERROR | QUAL | N | A | 3 | D | X | — |
| | QUAN | N | A | 3 | D | X | — |
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| | | | | | | | |
| ELEV ERROR | QUAL | — | — | — | U | — | — |
| | QUAN | N P | A C | 1 | N | X | P I |
| AZIMUTH ERROR | QUAL | — | — | — | U | — | — |
| | QUAN | N P | A C | 1 | N | X | P I |
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| THRUST ERROR | QUAL | — | — | — | — | — | — |
| | QUAN | — | — | — | — | — | — |
| | | | | | | | |
| ATTITUDE DIR | | — | — | — | — | — | — |
| THRUST D.R. | | — | — | — | — | — | — |
| PATH DIR | QUAN | A N P | C | 1 | N | X | P S I |
| SPEED DIR | QUAN | A N P | C | 1 | N | X | P S I |
| VERT VEL. DIR | | — | — | — | — | — | — |

42

# TABLE 3-3.6 INFORMATION ANALYSIS SUMMARY

2249 LUNAR LANDING — LETDOWN MODE

| INFO | TYPE | TASK (A,N,P,E) | PURPOSE A,C | FREQ OF USE 1,2,3 | IMPOR- TANCE N,D,U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P,A,S,I |
|------|------|------|------|------|------|------|------|
| ROLL | QUAL | A | A | 2 | N | X | A I |
|  | QUAN | A | A | 2 | N | X | — |
| PITCH | QUAL | A | A | 2 | N | X | A I |
|  | QUAN | A | A | 2 | N | X | — |
| YAW | QUAL | A | A | 2 | N | X | A I |
|  | QUAN | A | A | 2 | N | X | — |
|  |  |  |  |  |  |  |  |
| FPA | QUAL | — | — | — | U | — | — |
|  | QUAN | — | — | — | U | — | — |
| TOTAL VELOCITY | QUAL | — | — | — | U | — | — |
|  | QUAN | — | — | — | U | — | — |
|  |  |  |  |  |  |  |  |
| TARGET ELEV | QUAL | — | — | — | U | — | — |
|  | QUAN | N P | C | 1 | N | X | P I |
| TARGET AZIMUTH | QUAL | — | — | — | U | — | — |
|  | QUAN | N P | C | 1 | N | X | P I |
| TARGET RANGE | QUAL | — | — | — | U | — | — |
|  | QUAN | — | — | — | U | — | — |
|  |  |  |  |  |  |  |  |
| X DISPMNT | QUAL | — | — | — | U | — | — |
|  | QUAN | N | A | 3 | D | X | — |
| Y DISPMNT | QUAL | — | — | — | U | — | — |
|  | QUAN | N | A | 3 | D | X | — |
| Z DISPMNT | QUAL | — | — | — | U | — | — |
|  | QUAN | N | A | 3 | D | X | — |
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| MAIN THRUST | QUAL | N P | A | 2 | N | X | — |
|  | QUAN | N P | A | 2 | N | X | — |
| FUEL AVAIL | QUAL | ANPE | A | 1 | N | X | S |
|  | QUAN | ANPE | A | 1 | N | X | — |
|  |  |  |  |  |  |  |  |
| ORBITAL TIME | QUAL | — | — | — | U | — | — |
|  | QUAN | — | — | — | U | — | — |
| EVENT ELAPSED TIME | QUAL | N P | A | 2 | N | X | S |
|  | QUAN | N P | A | 2 | N | X | — |
| GREENWICH TIME | QUAL | — | — | — | — | — | — |
|  | QUAN | — | — | — | U | — | — |
|  |  |  |  |  |  |  |  |

| INFO | TYPE | TASK ANPE | PURPOSE A, C | FREQ OF USE 1 2 3 | IMPORTANCE N Q U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA |
| --- | --- | --- | --- | --- | --- | --- | --- |
| | | | | | | | P, A, S, I |
| ROLL RATE | QUAL | A | A | 3 | D | X | — |
| | QUAN | A | A | 3 | N | X | — |
| PITCH RATE | QUAL | A | A | 3 | D | X | — |
| | QUAN | A | A | 3 | N | X | — |
| YAW RATE | QUAL | A | A | 3 | D | X | — |
| | QUAN | A | A | 3 | N | X | — |
| | | | | | | | |
| FPA RATE | QUAL | — | — | — | U | — | — |
| | QUAN | — | — | — | U | — | — |
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| TGT ELEV RATE | QUAL | — | — | — | U | — | — |
| | QUAN | — | — | — | U | — | — |
| TGT AZ RATE | QUAL | — | — | — | U | — | — |
| | QUAN | — | — | — | U | — | — |
| TGT RGE RATE | QUAL | — | — | — | U | — | — |
| | QUAN | — | — | — | U | — | — |
| | | | | | | | |
| X VELOCITY | QUAL | — | — | — | — | — | — |
| | QUAN | A N | A C | 1 | N | X | — |
| Y VELOCITY | QUAL | — | — | — | — | — | — |
| | QUAN | A N | A C | 1 | N | X | — |
| Z VELOCITY | QUAL | — | — | — | — | — | — |
| | QUAN | A N | A C | 1 | N | X | — |
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| FUEL RATE | QUAL | A N P | A | 1 | N | X | — |
| | QUAN | A N P | A | 2 | N | X | — |
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| INFO | TYPE | TASK A N P E | PURPOSE A,C | FREQ OF USE 1 2 3 | IMPOR-TANCE N D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P,A,S,I | |
|------|------|------|------|------|------|------|------|------|
| ROLL ACCEL | QUAL | A | A | 3 | D | X | — | |
| | QUAN | — | — | — | — | — | — | |
| PITCH ACCEL | QUAL | A | A | 3 | D | X | — | |
| | QUAN | — | — | — | — | — | — | |
| YAW ACCEL | QUAL | A | A | 3 | D | X | — | |
| | QUAN | — | — | — | — | — | — | |
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| VEH X ACCEL | QUAL | — | — | — | U | — | — | |
| | QUAN | E | A | 2 | N | X | — | |
| VEH Y ACCEL | QUAL | — | — | — | U | — | — | |
| | QUAN | E | A | 2 | N | X | — | |
| VEH Z ACCEL | QUAL | — | — | — | U | — | — | |
| | QUAN | E | A | 2 | N | X | — | |
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| INFO | TYPE | TASK A, N, P E | PURPOSE A, C | FREQ. OF USE 1 2 3 | IMPOR-TANCE N, D U | PANEL DISPLAY | PROJECTED DISPLAY & CRITERIA P, A, S, I | |
|---|---|---|---|---|---|---|---|---|
| | | | | | | | | |
| ROLL ERROR | QUAL | A N | A C | 1 | N | X | — | |
| | QUAN | A N | A C | 1 | N | X | A I | |
| PITCH ERROR | QUAL | A N | A C | 1 | N | X | — | |
| | QUAN | A N | A C | 1 | N | X | A I | |
| YAW ERROR | QUAL | A N | A C | 1 | N | X | — | |
| | QUAN | A N | A C | 1 | N | X | A I | |
| | | | | | | | | |
| FPA ERROR | QUAL | — | — | — | U | — | — | |
| | QUAN | — | — | — | U | — | — | |
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| ELEV ERROR | QUAL | — | — | — | U | — | — | |
| | QUAN | N P | A C | 1 | N | X | P I | |
| AZIMUTH ERROR | QUAL | — | — | — | U | — | — | |
| | QUAN | N P | A C | 1 | N | X | P I | |
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| THRUST ERROR | QUAL | — | — | — | U | — | — | |
| | QUAN | N P | A | 3 | D | X | — | |
| | | | | | | | | |
| ATTITUDE DIR | | — | — | — | — | — | — | |
| THRUST DIR | | — | — | — | — | — | — | |
| PATH DIR | | — | — | — | — | — | — | |
| SPEED DIR | | — | — | — | — | — | — | |
| FT VEL DIR | QUAN | N P | C | 1 | N | X | PSI | |

technique to spacecraft operations. Extrapolation, from the sample cases presented, will allow evaluation of application of the technique to other specific cases the reader may be interested in.

The following information applies to the discussion and illustrations of the recommended displays in the succeeding paragraphs:

(1) "Real world" nomenclature refers to the actual object (whether it is the horizon, the aimpoint, or touchdown area) seen by the pilot through the window, as differentiated from the display system projected cues for these objects.

(2) In the case of lunar landing, the real world horizon, aimpoint, and touchdown area refer to the actual moon horizon, aimpoint and touchdown area on the moon surface.

(3) The "real world" (real moon) objects of interest are shown as dashed lines in the illustrations of the displays to differentiate them from the projected images.

(4) Most of the displays have been shown in the "non-zero" condition (i.e., with all commands not executed) to clearly identify the separate cues. The operation description of each mode indicates the display appearance when all commands have been executed and satisfied.

(5) For simplicity of drawing preparation, all displays are shown for a zero roll condition of the spacecraft.

(6) Control of switching between modes must be reviewed on a total system basis along with the controls for the panel displays and the spacecraft central computer. The switching may be manual or automatic, as required by system operation. Detailed mode switching information has not been included.

(7) Abbreviations are used in the drawings to avoid clutter, while at the same time permitting a maximum of descriptive nomenclature. Some abbreviations used are: DIR for director, ATT for attitude, TGT for target, BRG for bearings, LOS for line of sight, PERP for perpendicular, TD for touchdown, FOV for field of view, VFR for visual flight and IFR for instrument flight rules.

(8) The field of view (FOV) of the projected display is centered laterally and positioned vertically in the window FOV at a point which promotes maximum and optimum coverage. This vertical position may not, in some cases, be the spacecraft reference axis, since this location may not be optimum for the specific spacecraft type.

(9) The display field of view is shown as circular in the drawings. In practice, the exact shape of the field of view is dependent on the particular installation and the optical system used.

(10) The drawings show a solid line encompassing the field of view. This is for convenience of illustration only. No such line is required and none is used in the actual display.

## G. DISPLAYS FOR RENDEZVOUS

### 1. Review of Mission Mode Requirements

A spacecraft similar to the Gemini is the vehicle type used in this description of the rendezvous. The vehicle has a main thrust engine and a braking engine; thrust direction of the braking engine is opposite (180 degrees) to the thrust direction of the main engine. The center axis of the pilot normal field of view is approximately parallel to, and in the same direction as, the main engine thrust direction. Attitude and limited translational control is obtained by additional thrusters.

The rendezvous mission modes considered in this study are the initial (LOS rate arrestment) mode and the braking mode.

### a. Initial (LOS Rate Arrestment) Mode

(1) Objective - Maintain spacecraft on collision course by arresting any change in angular line of sight between the spacecraft and the target.

(2) Operation Required

● Orient spacecraft to reference pitch-yaw attitude.

● Thrust to command thrust value

### b. Braking Mode

(1) Objective - Control spacecraft-target range rate on the collision course so that range and range rate are reduced to zero at a preset range suitable for docking initiation.

(2) Operation Required

● Orient spacecraft in pitch and yaw along the LOS.

● Use main engine and braking engine to control range rate as a function of range.

● Monitor LOS rate and mode vernier corrections as required.

The braking mode is initiated after the LOS rate has been essentially reduced to zero; however, vernier corrections for LOS rates will probably be required.

In the descriptions of the braking mode operation, only that control action required for braking (i.e., range/range rate control) is discussed in detail. It is assumed that if the LOS rate is not zero, suitable LOS rate arrestment operation would be initiated.

    c.   Combined Mode

        (1)  Objective - Control spacecraft to control simultaneously LOS rate and range rate so that both are reduced to zero at a range suitable for docking initiation.

        (2)  Operation Required

           ● Orient spacecraft in pitch and yaw to programmed attitude

           ● Control thrust to obtain programmed range rate

  2.   Cue Form Description

The following is a description of the cue forms used in the displays for the rendezvous modes.

    a.   Spacecraft Reference Boresight

The location of the spacecraft boresight axis is indicated by a fixed circular image with straight line segments similar to wings on each side of the circular image and/or by indices at the top and sides of the display ( —O— ). The boresight indication, which is fixed with respect to the spacecraft, is required for logical interpretation of spacecraft attitudes relative to the LOS and the reference inertial plane.

    b.   Thrust Director

A small bar image, ( ⁻⁻—o ) moving at right angles to the left wing tip of the boresight image, indicates thrust command. The wing tip is the zero reference. Two fixed indices indicate scale reference points for thrust command magnitude. An acceleration command is indicated by an upward displacement of the image. The thrust command information is continuously and automatically computed for the mission phase and existing conditions.

    c.   Attitude Reference

The pitch-yaw reference attitude is indicated by a circular image ( O ) positioned at the reference attitude, when the reference attitude is with the FOV. When the reference attitude is outside the FOV, the image is automatically changed to a semicircle at the edge of the FOV.

Roll reference attitude, where required, is indicated by a two segment line which rotates about the boresight image and is positioned to the reference roll attitude.

In certain modes, where the required reference attitude is a specific plane and where a cue of that plane is being displayed, the usual reference roll attitude cue is not used, to eliminate display clutter.

d. Attitude Director

A two-line segment, cross-like image ( —+— ) is used for pitch-yaw attitude director control. Motion of the cue is vertical and lateral to the FOV in the plane perpendicular to the FOV.

The cue for pitch-yaw-roll attitude director control ( —+— ) is similar, but one line segment is shorter. Motion of this cue is the same as above for pitch-yaw director indications, but it also rotates to indicate roll attitude director information.

The director cue is of the "null" type with "fly-to" sensing and contains suitable damping control information for proper stability and performance. Operation technique is to "fly" the boresight image "to" the attitude director cue. When the proper attitude rates and accelerations are achieved, the boresight cue will overlay the attitude director cue.

e. Radar Target (Space Object)

The target position, in relative bearing form, is indicated by a circular image with a dot in the center when the target is within the field of view (FOV) of the projected display equipment. When the target is outside this FOV, the image is automatically changed to an arrowhead ( ↟ ). The arrowhead locates the plane of the boresight and target, and also indicates the direction in which to turn if it is desired to bring the target within the FOV. The target image position is controlled by information from the spacecraft radar system, and the target image may be made to vary in size as a function of range to the target.

f. Target LOS Rate

Target LOS rate is indicated by a line and arrow image. (●———▶) The line starts at the center of the space object target image (cue form e). The length of the line indicates the magnitude of the total LOS rate and the direction of the line and arrow indicates the direction of LOS rate.

g. Target LOS Perpendicular Plane

A line image indicates the plane which passes through the center of the spacecraft and is perpendicular to the target line of sight (LOS). The target LOS perpendicular plane serves as the reference pitch-yaw attitude plane during the initial (LOS rate arrestment) made.

h. Reference Inertial Target

A small image of fixed size and square form ( ☐ ) indicates a reference inertial target. This cue is initially automatically slewed to overlay the space object target (radar) and then is inertially stabilized. Change of direction of the line of sight between spacecraft and target may then be detected by displacement of the space object target (radar) image from the reference inertial target image.

The reference inertial target image size also serves as a scale reference when the space object target image size changes with range to target.

### i. Reference Inertial Plane

An image of three line segments indicates a plane which passes through the center of the spacecraft and is perpendicular to the Reference Inertial Target (cue form h). Displacement or motion of the target LOS perpendicular plane (cue form g) relative to this reference inertial plane indicates that the target LOS has changed or is changing.

### j. Elapsed Range and Range-To-Go

Qualitative range information is presented in symbolic form when the range reaches a preset value in the braking mode. A bar image, moving in a circular path at the bottom edge of the display, changes length as range changes. Two fixed indices indicate the start range point and the zero range point.

### k. Reference Range

The location of a small solid circular image, between the "start" and "zero range" indices of the range-to-go cue, gives the reference range value for the present time. The position of the reference range cue relative to the elapsed range cue provides qualitative information of range rate error.

## 3. Initial (LOS Rate Arrestment) Mode

In this mode, the pilot attains a collision course with the target by arresting the LOS rate. He does this by orienting the spacecraft in an attitude which is in a plane perpendicular to the LOS and at a specific attitude in this plane as determined by the magnitude and direction of the LOS rate. He then thrusts to obtain the required impulse in this attitude.

The correct pitch-yaw attitude may be attained by following the director commands until the reference attitude is attained. At this point the thrust command is used to obtain the necessary velocity impulse to arrest the LOS rate. The required impulse has been obtained when there is no longer a thrust command and the reference attitude is no longer moving relative to the inertially stabilized cues. The pilot may then use the target direction arrow, or the director to reorient the spacecraft to the LOS direction for initiation of the braking phase.

In the event no director or reference attitude cues are available, the pilot may use the radar or visual target location and the attitude display to arrest the LOS rate, although with a decrease in precision.

The astronaut observes the LOS rate direction with reference to a star or local body reference. The boresight is aligned with the target and a roll attitude achieved so that either a 90 degree pitch or yaw will align the thrust axis with the relative LOS rate vector. After a 90 degree pitch (or yaw, as appropriate) maneuver, thrust is applied. The amount of thrust (if there is

no thrust director) must be estimated by reference to radar panel indications, or the observed LOS rate. Following the thrust application, the reverse 90 degree pitch (or yaw) maneuver is made to acquire the target again and check the actual LOS rate.

Since flight control in this mode requires spacecraft orientation 90 degrees to the LOS and since spacecraft orientation may be, at times, any attitude relative to the LOS and to the flight path, it is necessary to review the display under the following conditions:

- Spacecraft attitude so that target bearing is within the field of view (FOV).

- Spacecraft attitude so that target line of sight perpendicular plane bearing is within the field of view.

- Spacecraft attitude so that target bearing and target line-of-sight perpendicular plane bearing are not within the field of view.

a. System 1 Display Operation For Initial (LOS Rate Arrestment) Mode

The display for this mode is shown in figure 3-3.1.

(1) Assessment and Control Information

The attitude director cue is used for control of attitude. The thrust director cue is used to control engine thrust. Assessment information must be obtained from the real world and the panel display.

(2) Operation

- The control action is to zero the boresight cue on the attitude director cue which will result in proper pitch-yaw rate/accelerations to achieve and maintain the reference attitude.

- When reference attitude is reached, use thrust director to control engine thrust to obtain the required impulse.

b. System 1 Display - Alternate Method - Initial Mode

(1) Assessment and Control Information

In the event that it is desired, and with the necessary switching of signal sources, the attitude director cue may be used as a reference attitude cue for achieving the proper attitude for thrusting.

(2) Operation

- Control spacecraft attitude by manually establishing pitch-yaw rates to align reference attitude with boresight.

- When reference attitude is reached, use thrust director to control engine thrust to obtain the required impulse.

c. System 2 Display For Initial (LOS Rate Arrestment) Mode

The display for this mode is shown in figure 3-3.2.

If spacecraft attitude is such that the target bearing is within the FOV, the following applies (see figure 3-3.2):

(1) Assessment Information

- Radar target cue indicates present position of target.

- Target LOS rate cue indicates direction and magnitude of the LOS rate.

- Reference inertial target cue indicates original position of target.

- Reference inertial target cue position relative to radar target cue indicates magnitude and direction the LOS has moved after the initial lock on.

(2) Control Information

- Attitude director cue furnishes quickened pitch-yaw attitude control information to achieve and maintain the reference attitude.

- Thrust director cue indicates thrust change required to achieve and maintain the command impulse.

If spacecraft attitude is such that the target LOS perpendicular plane is within the FOV, the following applies (see figure 3-3.2b):

(1) Assessment Information

- Target arrow indicates direction of radar target.

- Target LOS perpendicular plane (1) indicates the plane in which the attitude reference will occur.

- Reference inertial plane (2) indicates the original (at lock on) plane of the target perpendicular plane.

- Lateral and/or rotational displacement of plane (1) relative to plane (2) indicates that the target has changed from its original position.

- Attitude reference indicates required pitch-yaw attitude.

(2) Control Information

- Attitude director cue furnishes quickened pitch-yaw attitude control information to achieve and maintain the reference attitude.

- Thrust director cue indicates thrust change required to achieve and maintain the command impulse.

NOTE: Attitude reference would be control as well as assessment information if the attitude director were not used.

If spacecraft attitude is such that neither the target bearing nor the target LOS perpendicular plane is within the FOV, the following applies (see figure 3-3.2c):

(1) Assessment Information

- Target arrow indicates direction of target.

(2) Control Information

- Attitude director cue furnishes quickened pitch-yaw attitude control.

- Thrust director cue indicates thrust change required to achieve and maintain the command impulse.

Operation is as follows:

Control attitude to align the boresight with the pitch-yaw attitude director cue. This will result in proper pitch-yaw attitude rates/accelerations to achieve and maintain the reference attitude.

Thrust as required by the thrust director.

There is no specific requirement on roll attitude. The spacecraft may be in any roll attitude and still be properly oriented if the pitch-yaw reference attitude is attained. Figure 3-3.3A shows the display when the reference pitch-yaw attitude is reached and the thrust director calls for a velocity impulse.

In actual practice, the pilot would maintain an attitude in, or close to, the LOS perpendicular plane, even if there was no LOS change at the time.

(a) If he had elected to keep the target directly opposite his left or right shoulder, when the spacecraft is oriented in the target LOS perpendicular plane, then only direct pitch control is required to attain the reference attitude when thrusting is required due to an LOS change. See figure 3-3.3B for display.

(b) If he had elected to keep the target directly in line with his seat-head axis, when the spacecraft is oriented in the target LOS perpendicular plane, then only direct yaw control is required to attain the reference attitude when thrusting is required due to an LOS change. See figure 3-3.3C for display.

After the proper thrust has been applied, and the LOS change becomes zero, or below a preset acceptable magnitude, the following occur:

- The attitude director is automatically deactivated to the zero position, i.e., at the boresight.

- The thrust director calls for zero thrust.

- The reference attitude is automatically deactivated and held at the last pitch-yaw reference attitude in the target LOS perpendicular plane. The reference attitude symbol is changed to a diamond form under this condition.

The commands are automatically activated if the LOS rate increases above the preset magnitude.

In addition, automatic initial alignment of the reference inertial target with the radar target (and consequently the reference inertial plane to the target LOS perpendicular plane) takes place when the commands are deactivated.

Figures 3-3.4A, B, C show the display after the LOS change has been arrested. (Spacecraft orientation is at various attitudes similar to figure 3-3.2.)

4. Braking Mode

After the LOS rate is arrested and the vehicle is on a collision course with the target, the braking mode is initated. Braking action to reduce the relative spacecraft-target velocity to zero at some preset range, suitable for docking initiation, is required.

The pilot is required to orient the spacecraft thrust axis in pitch and yaw along the LOS and apply thrust to control range rate as a function of range. Vernier LOS rate correction will be required to maintain a collision course.

The radar target cue, or the actual target, if visible, permit precise thrust axis alignment with the LOS for thrust application. The radar target cue size growth with decreasing range (or the real target growth) and the fixed boresight cue size permit qualitative range monitoring. The elapsed range cue furnishes more quantitative range data throughout this phase. The reference range cue indicates whether the closure rate is correct by showing the programmed range corresponding to the present range rate. Thus, if the reference range dot is ahead of the range bar, the present velocity is too slow. The astronaut may accelerate until the two cues align, or conserve fuel and wait until the range bar catches up to the reference range dot (i.e., the range decreases to the value appropriate to the current velocity).

The thrust director furnishes continuous thrust commands according to the range/range rate program. The range-to-go cue provides a sensitive assessment display.

Since thrust is applied in a LOS direction, orbital mechanics effects will cause LOS rate errors. The observed LOS rates are easily controlled with the heads up display. The direction of the LOS rate is observed relative to the

spacecraft axes, and the astronaut can roll the vehicle to align either set of translational thrusters (i.e. vertical or lateral) in the direction of the velocity error prior to thrusting. Alternatively, the components of the LOS rates in the present vehicle axes directions may be zeroed by combinations of translational thrusts without roll.

A projected display is also convenient for use of the braking thrust for simultaneous LOS control. If while braking, a LOS rate develops, the thrust axis may be inclined opposite the LOS rate until the LOS rate is zeroed. The effect on the braking thrust component will be insignificant for the small tilt angles required.

a. System 1 Display For Breaking Mode

The display for this mode is the same as that used in the initial mode (see figure 3-3. 1).

Operation is identical except that now the reference attitude is along the LOS, and the thrust director will be used primarily for braking (i.e., reduction of range rate).

Assessment information must be obtained from the real world view and the panel display.

b. System 2 Display For Braking Mode

The recommended display for this mode is shown in figure 3-3. 5. The display is identical to the display for the initial mode except for the following:

- The reference attitude is now along the LOS.

- The thrust director is now a function of range, range rate and range acceleration.

- The inertial reference plane and the target LOS perpendicular plane are not used, since the control action used (i.e., thrust control) is along the LOS.

- The radar target cue size is changed as a function of range. Thus, when its size becomes equal to the reference inertial target, (or the boresight arc) a preset range has been reached.

- Range to go, in qualitative form, is displayed when the range is below a fixed preset value.

- Reference range, in qualitative form, is displayed as the range which should exist for the current range rate. It is thus a monitor on the range rate profile.

If the spacecraft attitude is such that the LOS is in view, the following applies (see figure 3-3. 5A):

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(a) Assessment Information

- Radar target indicates present position of target LOS, which is now the attitude reference.

- Target LOS rate cue indicates direction and magnitude of the LOS rate.

- Reference inertial target indicates original position of target.

- Reference inertial target relative to radar target gives a measure of the magnitude and direction of the LOS change.

- Target size relative to reference inertial target (or boresight cue) size is a measure of range (i.e., target size will equal reference target size at a preset range and size increases with decreasing range).

- Elapsed range and range to go provides qualitative distance information.

- Reference range for present time, relative to elapsed range, provides a qualitative indication of range rate error.

(b) Control Information

- Attitude director cue furnishes quickened pitch-yaw attitude control information.

- Thrust director furnishes velocity impulse commands as a function of range/rate/acceleration.

If the spacecraft attitude is such that the LOS is not in view, the following applies (see figure 3-3.5B):

(a) Assessment Information

- Target arrow indicates direction of target (LOS and reference attitude).

- Elapsed range and range to go provides qualitative distance information.

- Reference range for present time, relative to elapsed range, provides a qualitative indication of range rate error.

(b) Control Information

- Attitude director cue furnishes quickened pitch-yaw attitude control information.

- Thrust director indicates thrust commands. Boresight (thrust) axis must be aligned with LOS before applying thrust.

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Operation is as follows:

Control attitude to align the boresight with the pitch-yaw attitude director cue. This will result in proper pitch-yaw rates/accelerations to achieve and maintain the reference attitude, which is the LOS. The reference attitude is reached when the radar target cue is centered in the boresight cue.

Adjust thrust as required by thrust director. Proper velocity is achieved when the director bar is zeroed at the boresight wing tip. Proper range rate-range situation is achieved and indicated when the reference range position becomes aligned to the end of the elapsed range bar. Note that there is no specific requirement on roll attitude. The spacecraft may be in any roll attitude and still be properly oriented if the pitch-yaw attitude reference is attained.

5. Combined Initial (LOS Rate Arrestment) and Braking Mode

The previous pages have described application of the projected display to the rendezvous technique which uses separate initial (LOS rate arrestment) and braking modes. Appendix C describes in detail a rendezvous technique which combines the tasks of LOS rate arrestment and braking into one mode.

The operation required is as follows:

● Orient the spacecraft to reference attitude.

● Continuously thrust to thrust command.

(Reference attitude and command thrust are determined by computer and are a function of the LOS, the LOS rate, the range and the range rate).

In practice, correction of LOS rate is rapid. Thus, the remaining control operation is primarily one of braking with small corrections of LOS rate.

The previous mode descriptions are generally applicable to this mode for situations using attitude and thrust director operation. The range display and the inertially stabilized cues furnish necessary assessment information during the maneuver. If at any time the astronaut is dissatisfied with the progress of the maneuver, due to computer malfunction, for example, he may easily switch to the LOS arrestment and braking modes previously described with minimum transition difficulty.

a. System 1 Display For Combined Initial and Braking Mode

Display and operation is the same as that used for the initial and braking modes. See figure 3-3.1. (The reference attitude and the command thrust are determined by a central computer.)

Assessment information must be obtained from the real world view and the panel display.

b. System 2 Display For Combined Initial and Braking Mode

The recommended display for this mode is shown in figure 3-3.6. (The display is similar in appearance to that of the braking mode figure 3-3.5 except that the reference attitude and command thrust are those required for combined LOS rate arrestment and braking.)

If the spacecraft attitude is such that the target bearing is within the FOV but the reference attitude is not in the FOV, the following applies (see figure 3-3.6A):

(1) Assessment Information

- Radar target indicates present position of target and LOS.

- Target LOS rate cue indicates direction and magnitude of LOS rate.

- Reference inertial target cue indicates original position of target.

- Reference inertial target cue position relative to radar target cue indicates magnitude and direction the LOS has moved after initial lock on.

- Attitude reference cue indicates pitch-yaw attitude reference.

- Elapsed range and range to go provides qualitative distance information.

- Reference range for present time, relative to elapsed range, provides qualitative indication of range rate error.

(2) Control Information

- Attitude director furnishes quickened pitch-yaw attitude control information.

- Thrust director furnishes thrust command information.

If the spacecraft attitude is such that the target is not in view, the following applies (see figure 3-3.6B):

(1) Assessment Information

- Target bearing cue indicates target direction.

- Reference attitude indicates reference pitch-yaw attitude if in FOV, or the direction of the reference pitch-yaw attitude if not in FOV.

- Elapsed range and range to go provides qualitative distance information.

- Reference range for present time, relative to elapsed range, provides qualitative indication of range rate error.

(2) Control Information

- Attitude director furnishes quickened pitch-yaw attitude control information.

- Thrust director furnishes thrust control information.

Operation is as follows:

Control attitude to align the boresight with the pitch-yaw attitude director cue. This will result in proper pitch-yaw rate/accelerations to achieve and maintain the reference attitude. (The reference attitude is reached when the boresight cue overlaps the reference attitude cue.)

Adjust thrust as required by thrust director cue. Proper arrestment of the LOS rate is indicated when the LOS rate cue is zero. Proper range rate-range situation is achieved and indicated when the reference range position becomes aligned to the end of the elapsed range bar. Note that there is no specific requirement on roll attitude. The spacecraft may be in any roll attitude and still be properly oriented if the pitch-yaw attitude reference is attained.

6. Advantages of the Projected Display System for Rendezvous Operations

The use of the projected display system for rendezvous has the following advantages:

- provides a heads-up display in the pilot's normal field of vision.

- provides information of three-dimensional context in the best understood form with a 1:1 scale and with compatible sensing with the real world.

- presents synthetic visual information for use under "instrument flight" conditions, such as when the range to target is great and the target size is too small to be seen, or when the required attitude for thrusting does not permit viewing the target directly.

- provides both assessment and control display information in the pilot's forward external view to improve visual flight control.

- allows common display/control techniques for both "visual flight" and "instrument flight" conditions.

The projected display system specifically allows the pilot, while observing the normal external view to:

- Locate rapidly the target by use of the target cue.

- Determine quantitative LOS changes by use of the LOS rate cue.

- Determine qualitatively LOS changes by use of the reference inertial target cue.

- Determine simply and directly the proper roll attitude so final attitude change required for thrusting is either a pitch change or a yaw change as desired (in the initial mode).

- Assess rapidly the present attitude relative to proper pitch-yaw attitude for thrusting.

- Make vernier LOS rate correction by simple alignment of the braking thrust vector away from the LOS, opposite the direction of the LOS rate (in the braking mode).

- Obtain range determination by growth of target cue size relative to fixed size of the inertial reference cue, or the boresight cue.

- Assess rapidly the range rate and range through use of the reference range cue and the elapsed range cue.

- Obtain effective attitude control through use of the attitude director cue, which contains any necessary "quickening" information.

- Obtain rapid thrust control by use of the thrust director cue which contains any necessary "quickening" information.

- Make maximum use of familiar aircraft "formation flying" experience.

## H. DISPLAYS FOR LUNAR LANDING

### 1. Review of Mission Requirements

The lunar landing mission consists of three modes:

- Initial Descent - descent from orbit to start of approach
- Approach - approach to hover prior to start of letdown
- Letdown - final letdown to touchdown

Figures 3-3.7 is a typical lunar landing flight profile.

#### a. Initial Descent Mode

##### (1) Objective

The objective of the initial descent mode is to transfer the spacecraft from an elliptical lunar orbit in a programmed maneuver (automatic or manual control) while preparing for the approach and landing. In accomplishing this objective, the pilot must be able to observe the lunar surface, monitor system performance and continually evaluate his progress for "Go or No-Go" indications.

##### (2) Operation Required

In the initial descent mode the following operations are required:

- Orient spacecraft thrust axis in a direction 180 degrees from the velocity vector
- Orient spacecraft yaw axis so window field of view (FOV) is in the orbital plane and aligned to view the moon surface
- Thrust to reduce velocity, according to program.
- At proper altitude, make 180 degree yaw change to permit continued view of landing area prior to start of approach.

#### b. Approach Mode

##### (1) Objective

The objective of the approach mode is to approach the intended area on a stable, controlled path with programmed decrease in vertical and translational velocities.

(2) Operation Required

- Orient spacecraft in the direction of selected landing area

- Use thrust and attitude control to achieve desired flight path to landing area

- Use thrust and attitude control to reduce forward and vertical velocities to zero, or small values, for final landing site inspection.

c.  Letdown Mode

(1)  Objective

The objective of the letdown mode is to maneuver the spacecraft from the hover point to a safe landing on the lunar surface.

(2)  Operation Required

- Control thrust to descent to lunar surface with a safe touch-down rate of descent

- Maintain spacecraft translational velocities within safe limits through tight attitude control

- Maintain spacecraft attitude within safe limits at touchdown.

2.  Cue Form Description

The following is a description of the cue forms used in the displays for the lunar landing modes.  (The asterisk denotes cues previously used and described for other modes.  The description of these cues is repeated here in brief form for completeness of this section.)

a.  Fuselage Reference (Boresight)*

The location of the spacecraft reference (boresight) axis is indicated by a fixed circular image with straight line segments similar to "wings" on each side of the circular image and/or by indices at the top and sides of the display.  (—      —O—      —)  The boresight image is used for logical interpretation of spacecraft attitude relative to reference inertial planes and to the real moon horizon.

b.  Thrust Director*

A small bar image ( ⁻ ⁼O  ) moving relative to the left "wing" tip of the boresight image indicates thrust command.

The wing tip is the zero reference.  Two fixed indices indicate scale references for thrust command magnitude.  An acceleration command is indicated by an upward displacement of the image.  The thrust command information is automatically and continuously computed for the mission phase and existing conditions.

## c. Attitude Reference*

The pitch-yaw reference attitude is indicated by a circular image (O) positioned at the reference attitude when the reference attitude is within the FOV. When the reference attitude is outside the FOV, the image is automatically changed to a semi-circle at the edge of the FOV.

Toll reference attitude, where required, is indicated by a two-segment line which moves rotationally about the boresight image and is positioned to the reference roll attitude. In certain modes, where the required reference attitude is a specific plane and where a cue for that plane is being displayed, the usual reference roll attitude cue is not used, to eliminate display clutter.

## d. Attitude Director*

A two-line segment, cross-like image (+) is used for pitch-yaw-attitude director control. Motion of this cue is vertical and lateral in the FOV in the plane perpendicular to the FOV. The cue for pitch-yaw-roll attitude director control (→+—) is similar to that of the pitch-yaw-attitude director, but one line segment is shorter. Motion of this cue is the same as above for pitch-yaw director indication, but it also rotates to indicate roll attitude director information.

The director cue is of the "null" type with "fly-to" sensing, and contains suitable quickening information for proper stability and performance. Operation is to fly the boresight image "to" the attitude director cue. When the proper rates and acceleration are achieved, the boresight cue will overlay the attitude director cue.

## e. Speed Director

In the approach mode, a small circular image ( ●  ═══─┐  ), moving at right angles to the left "wing" tip of the path control director image, is used for a speed director cue.

The wing tip is the scale reference zero point. Two fixed indices indicate scale references for the speed director cue. The speed director information is automatically and continuously computed for the mission phase and the existing conditions at the time. The speed director cue and the thrust director (cue b) are never used in the same mode.

## f. Orbital Plane

This cue is a line which is aligned parallel to and within the orbital plane. A modified form of this cue is a short bar line on the horizon cue.

It is used with the boresight image to show spacecraft attitude relative to the orbital plane.

g. Nadir and Zenith

The Nadir and Zenith attitude points are marked by a short bar symbol. These cues are used with the orbital plane cue and with the all-attitude pitch scale cue.

h. Local Horizon

A line image represents the true horizontal plane in the real world. This image overlays the real moon horizon at low altitudes and is displaced above the real moon horizon by the dip angle at higher altitudes. When used with the boresight cue, it furnishes pitch and roll information.

i. Velocity Vector Perpendicular Plane

A three segment line image (— —+— —) indicates the plane which passes through the center of the spacecraft and is perpendicular to the direction of the spacecraft velocity vector.

The velocity vector perpendicular plane cue is used in the initial descent mode. Control of the spacecraft attitude, using this cue as the reference plane, results in the proper thrust alignment 180 degrees from the direction of the velocity vector.

j. All Attitude Pitch Scale

A pitch scale (⊩⎴⎴⎴⎴⎴⎴⎴⎴⎴⎴⎴⎴) with 10, 30, 50, 70, degree reference points is used for pitch indications and is of particular importance when the horizon reference is out of the field of view. It also furnishes qualitative roll information.

The scale remains laterally centered in the FOV.

k. Selected Ground Tracking Heading

The heading of the selected ground track is indicated by a short line perpendicular to the horizon ( ——+—— ). The location of this heading in the real world is indicated by this image. The location of the boresight relative to this image indicates the relative heading difference.

l. Deviation Marker

This cue ( ◯ ), of elliptical form, indicates displacement deviation from the desired path in the vertical and lateral directions during approach. The cue is gyro stabilized below the horizon at the desired glide path angle and laterally stabilized in the reference heading (selected ground track) direction.

The location of the aim point (either the real moon aimpoint or the aimpoint cue) relative to the center of the ellipse indicates off-course angular displacements.

During the letdown mode, the ellipse remains stabilized with heading. It is positioned vertically below the horizon as a function of altitude and the distance $X_0$ (figure 3-3.12). The longitundinal displacement of ellipse center from the aimpoint is, therefore, a direct indication of longitudinal displacement from the desired letdown position on the ground. Similarly its lateral displacement indicates lateral positional error.

### m. Path Control Director

An image with a form of 2 wings ( ⌐ • ⌐ ) is used for a path control director cue. Control of the spacecraft to keep this director cue aligned with the aim point, either the real world aimpoint or the aimpoint cue (item p), results in path control to attain and maintain a selected glide path and a selected ground track. Operating technique is to fly the director cue to the aim point.

Motion of the cue is vertical and lateral in the plane perpendicular to the FOV. The angular position of its "wings" relative to the horizon indicates roll attitude. The path director cue includes quickened information as required.

### n. Vertical Velocity Director

A small diamong image ( ◇ ), moving at right angles to the horizon, is used in the letdown mode to command vertical rate as a function of altitude to a specified low rate at touchdown. Thrust is adjusted to maintain the vertical velocity director cue superimposed over the aimpoint. The depression angle of this image from the horizon is equal to the sight line angle to the aimpoint, plus the altitude rate error, plus any necessary quickening, and a bias to account for eye-to-vehicle landing gear height.

The maximum altitude rate command is limited as a function of altitude, independent of the sight line angle to the aimpoint, to insure a proper small altitude rate at touchdown regardless of the longitudinal or lateral position errors at touchdown.

The vertical velocity director cue is laterally stabilized at the selected ground track heading. It may, if desired, be stabilized directly at the aimpoint lateral bearing angle and thus always move in a vertical line over the aimpoint.

### o. Altitude Scale

The approach and letdown modes require precise data on absolute altitude above the moon surface. Thus an altitude scale is added to the altitude display. Actual altitude above the moon surface is read where the horizon cue intersects the altitude scale. The altitude scale is consistent pictorially with a tower rising from the ground with altitude markers on it.

If the spacecraft attitude is such that the horizon is not in view, the altitude scale will not be readable or may be totally out of view. In this case a numeric symbolic image is used at the right center edge of the display for altitude information.

### p. Aimpoint

The aimpoint cue ( Δ ) is a triangular-shaped image which is the termination of the desired glide path on the ground. It overlays the actual aimpoint on the moon. It is the reference for the off-course deviation indication (deviation ellipse) and for the letdown vertical rate control when the real world aimpoint is obscured. A modified form of the cue (i.e., a short bar line ( — ) may be used if the selected ground track cue (r) is a part of the display.

### q. Touchdown Area

The touchdown area cue ( x ) is automatically positioned to always overlay the real moon touchdown area.

### r. Selected Ground Track

A straight line image is used to indicate the selected ground track. The line image will start at the selected ground track heading cue ( cue k) on the horizon and extend, i.e., be superimposed, over the real world aim point and touchdown area.

The selected ground track cue provides: more rapid identification of the real world aimpoint and touchdown area; computer monitoring since the ground track cue should be directly superimposed over the real world points; and more graphic indication of relative heading of the spacecraft to the selected track.

### s. Fuel Remaining

Qualitative fuel remaining is presented in symbolic form. A bar image, moving in a circular path at the bottom edge of the display, changes length as fuel remaining changes. Two fixed indices indicate the start and zero fuel remaining conditions.

This cue is used in the approach and letdown modes where allowable fuel is limited. An alternate method would be to use the cue for time to go.

### t. Perspective Touchdown Area

A touchdown area cue with a form that changes shape and size with bearing, range and attitude is used in System 3 display.

u.  Max Safe Range Footprint

A foot print type cue is used to indicate the maximum safe range remaining considering the vehicle's present altitude, velocities, and fuel remaining for the mode.

The prediction for the footprint size, position and shape may be based on: (1) a straight line descent path from the vehicle present position to the max range point, or (2) a constant altitude profile with lateral and longitudinal translation and then an immediate vertical descent as would probably be used in the letdown mode.

Any real world aimpoint may be safely selected provided it falls within the area of the footprint.

3.  Initial Descent Mode

a.  System 1 Display for Initial Descent

A System 1 display is not readily applicable to this mode, since pitch-yaw-roll and thrust control is required and there is no appropriate real world point to use as a reference. The System 1 display is limited, by definition, to a simple display with one index cue with two degrees of freedom, but not including rotational freedom, and a second index cue with one degree of freedom.

b.  System 2 Display (Initial Descent)

The display for System 2 is shown in figure 3-3.8 as it would appear for the spacecraft positions A, B, and C of the flight profile (see figure 3-3.7).

(1)  Assessment Information

● Orbital plane relative to boresight indicates spacecraft attitude displacement from orbital plane

● Orbital plane is roll attitude reference

● Nadir cue indicates pitch attitude reference at start of descent

● Reference attitude cue indicates computed pitch-yaw attitude reference. This cue will move along the orbital plane cue until time of position C at which point it will move along the horizon line for the 180 degree yaw turnaround command.

● Horizon cue provides indication of local horizon

● Boresight relative to horizon cue provides attitude information.

(2) Control Information

- Attitude director furnishes quickened pitch-roll-yaw attitude control information to achieve and maintain the required reference attitude

- Thrust director indicates thrust change (with quickening if required) to achieve and maintain the command thrust.

(3) Operation

Control attitude to continuously "fly" and align the boresight cue to the attitude director cue. This results in proper pitch-yaw-roll rates and accelerations to achieve and maintain the reference attitude at which time the boresight cue will overlay the reference attitude cues, i.e., the pitch-yaw attitude reference cue will be centered in the boresight cue, and the orbital plane cue will pass through the boresight cue normal to the "wings". (It is not necessary that the pitch-yaw command cue be centered in the boresight laterally if it is centered vertically, to attain proper thrust axis direction. However, the roll attitude must be 90 degrees to the orbital plane.)

Change thrust until the thrust director cue is zeroed, which indicates that the proper command thrust has been attained to maintain the desired velocity profile.

c. System 2 Alternate Display (Initial Descent)

An alternate display is shown in figure 3-3.9 as it would appear for the spacecraft positions A, B, C of the flight profile of figure 3-3.7.

(1) Assessment Information

- Velocity vector perpendicular plane provides pitch-roll attitude reference for spacecraft attitude. Spacecraft may be at any yaw attitude but still properly oriented if it is at the pitch-roll reference attitude.

- Orbital plane cue provides attitude information relative to the orbital plane

- Local horizon cue provides indication of local horizon

- Boresight relative to horizon cue provides attitude information

- Yaw attitude reference cue indicates yaw required (if desired) to maintain the spacecraft yaw in the orbital plane (Positions A thru C) and at C information on yaw required for the 180 degree turnaround.

- Velocity vector perpendicular plane relative to the horizon indicates the velocity vector direction relative to the local vertical

- Velocity vector perpendicular plane relative (i.e., parallel or non-parallel) to the horizon, at the orbital plane cue, indicates if the spacecraft yaw is in the orbital plane.

(2) Control Information

- Attitude director furnishes quickened pitch-roll attitude control information to achieve and maintain the required reference attitude during the time of travel from positions A to C. The use of quickened yaw attitude control information is optional since the spacecraft may be at any yaw attitude. At C however, the attitude director furnishes quickened pitch-roll-yaw attitude which includes the 180 degrees yaw turnaround requirement.

(3) Operation

Operation is as follows:

- Change attitude to continuously fly and align the boresight to the attitude director cue. The boresight will then overlap the velocity vector perpendicular plane which is the reference pitch-roll attitude for positions A to C. At C, the reference yaw attitude is changed 180 degrees.

- Change thrust until the thrust director cue is zeroed, which indicates that the proper thrust program is being followed.

4. Approach Mode and Letdown Mode

The approach and letdown modes are very closely related in operation and time and thus in assessment and control information requirements. These modes will be jointly discussed in this section.

a. Approach

The geometry for the approach mode is shown in figure 3-3.10. The selected track may be obtained in two ways. If at the approximate 1000-foot hover point, the pilot picks a landing area which he may descend to on a straight line from his present position, the selected track becomes that line connecting his present and aimpoint positions. If there is a glide path angle restriction to be considered, this glide path angle may be selected and the path control director will command a coupling maneuver to the selected path. Prior to initiation of the descent toward the aiming point, the path control image may be operated in a path direction mode in which the cue indicates the actual direction of the velocity vector in the visual field, thus simplifying the initial acquisition of the descent aimpoint for the subsequent director control.

The initial aimpoint and landing area selection is done visually by the pilot. At the start of the approach, these two areas are effectively the same. As the approach progresses, the piolet judges whether the area short of the aimpoint which he would descend to from final hover is suitable, and, if so, continues the approach. If closer observation indicates the landing area is unsuitable, translation to an acceptable area is required at a low attitude, and the flight path mode of operation for the path image may be used for control.

During the approach, the deviation cue indicates lateral and glide path displacement errors relative to the initially selected descent path. The path director furnishes continuous quickened path control information, and the speed control cue provides continuous quickened speed (rate of descent) control. The horizon and auxiliary attitude reference display furnishes a sensitive, precise reference for attitude control and monitoring.

b. Letdown

The geometry for the letdown made is shown in figure 3-3.11. The deviation ellipse is stabilized laterally with heading. It is positioned vertically below the horizon as a function of altitude and $X_0$ (the distance between the touchdown point and the aimpoint). The displacement of the ellipse center from the aimpoint is therefore a direct indication of lateral and longitudinal displacement from the desired letdown position on the ground.

The vertical velocity director image is used to command vertical rate as a function of altitude to a specified low rate at touchdown. Thrust is adjusted to maintain the vertical velocity director cue superimposed over the aimpoint. The depression angle of this image from the horizon is equal to the sight line angle, $\lambda$, plus an angle, $\beta$, which is proportional to the altitude rate error (from the reference altitude rate for the present altitude) plus any necessary quickening. A bias is also added to account for eye-to-landing strut foot height. An additional bias may be included so the reference zero altitude rate occurs at a 2- or 3-foot altitude rather than at zero feet in altitude. (See figure 3-3.11 A and B.)

If the vehicle moves from over the touchdown point, which would be indicated by the deviation cue, the reference altitude rate will be somewhat in error, i.e., it would be increased if the vehicle had moved forward longitudinally, (see figure 3-3.12 C) and reduced if the vehicle had moved back longitudinally. Altitude rate limiting would be used to avoid unsafe rates. An alternate method is to configure the system so the vertical velocity director is used relative to the deviation ellipse cue, thus avoiding the errors in the reference altitude rate information if the prescribed vertical path is not followed.

The above case is for use on a real (i.e., moon) aimpoint where no radar information relative to the aimpoint is available. $X_0$, used to determine $\lambda$, is a fixed preset value based on the anticipated distance between the touchdown point and the aimpoint for the typical approach and letdown.

Where a navigational radar or a beacon is used to locate the aimpoint, the aimpoint is located in the projected display in its correct real world position. Thus the vertical velocity director cue can always be used relative to the aimpoint, as the computer would adjust the reference altitude rate to compensate for any longitudinal displacements of the vehicle from the touchdown (and thus aimpoint) point.

c. System 1 Display for Approach and Letdown

Operation with Display System 1 is as follows (see figure 3-3.12). The System 1 display has limited capacity and flexibility and only the path control director and speed control director cues are available for use.

At the initiation of the approach mode, the pilot selects a general landing area. Using this as the aimpoint, he controls the spacecraft vertical and lateral path (as described above and in subparagraph 3-2) to maintain the path control director cue superimposed over the real world aimpoint. Speed control is achieved by use of the speed director cue. As altitude decreases and the vehicle approaches the aimpoint along the selected track, a final survey of the surface for a safe touchdown area occurs. If a touchdown area is selected short of the aimpoint, the approach is stopped over the selected touchdown area and the letdown is started. If the touchdown areas were beyond or to either side of the aimpoint, the approach is stopped at a safe altitude and the vehicle is translated to the touchdown area for the letdown.

The letdown mode display is shown in figure 3-3.12 B. Control is achieved by changing vertical velocity so the vertical velocity cue remains superimposed on the real world aimpoint. Assessment information must be obtained by the pilot from the real world view and the panel displays.

d. System 2 Display for Approach and Letdown

Operation with Display System 2 is as follows (see figure 3-3.13).

(1) Approach Mode

(a) Assessment Information

The assessment information is as follows:

- Horizon and pitch scale cues indicate spacecraft, pitch and roll attitude

- Deviation cue relative to real world aimpoint indicates vertical and lateral position displacement from selected descent path

- Selected ground track heading cue (on horizon) indicates relative heading of spacecraft to selected track

- Fuel remaining cue indicates fuel remaining for this maneuver

NOTE: System 2, by definition, does not include perspective cues. The lack of perspective cues does not permit assessment of closure rates and reduces the effectiveness of positional cues. System 3 includes the perspective information

(b) Control Information

● The control operation is as follows:

● Path control director is maintained superimposed on real world aimpoint for path control

● Speed director cue is used for proper speed control.

(c) Operation

Operation is as follows:

● Fly and continuously align the path control director cue to the real world aimpoint

● Continuously fly the speed control director to its zero reference.

This results in proper altitude control and thrust control to achieve and maintain the desired descent path to the aimpoint.

When the proper command action is satisfied, the path control director will overlay the real world aimpoint, the speed director will be zeroed and the deviation cue will gradually move and finally overlay the real world aimpoint as the lateral and vertical position displacement errors from the selected glide path and ground track are reduced to zero.

(2) Letdown Mode

(a) Assessment Information

The assessment information is as follows:

● Horizon and pitch scale cues indicate spacecraft pitch and roll attitude.

● Deviation cue relative to real world aimpoint is an indication of longitudinal and lateral displacement from the touchdown area

● Ground track heading cue (on horizon) indicates heading of spacecraft relative to selected track.

● Fuel remaining cue indicates fuel remaining for this maneuver.

(b) Control Information

The control information is as follows:

● Vertical velocity director cue location relative to the aimpoint is used to control vertical velocity

(c) Operation

Operation is as follows:

- Attitude is maintained at or near level to minimize any lateral accelerations due to thrust tilt

- Thrust is adjusted to maintain vertical velocity director cue superimposed over the real world aimpoint

- Pilot monitors lateral and forward velocity by noting position of the deviation cue relative to the aimpoint.

The above descriptions cover operation after the aimpoint, glide path and ground track to the aimpoint have been selected.

In applications where this selection is done manually, the following technique may be used. This technique may be used any time during the approach to rapidly change to a new aimpoint, assuming that:

- It is desired to select a glide path directly from the vehicle present position to the visually selected aimpoint

- It is desired that the selected aimpoint be directly in line with the vehicle velocity vector which would be the case if the pilot were maneuvering the vehicle along in a forward velocity direction and gradually descending while searching for a suitable landing area.

Pushing a button, changes the function of the path director cue to a flight path (velocity vector) cue and slews and slaves the deviation cue to this flight path cue. Visual control to align the flight path cue over an aimpoint means the vehicle velocity vector is in line with the aimpoint. Pushing the button again returns the system to normal "approach" mode operation. The deviation cue becomes inertially stabilized to the last aimpoint bearing that existed at the instant the button was pushed and the path cue again functions as a path director. A selected glide path and ground track from the vehicle present position to the aimpoint along the existing velocity vector direction has thus been established.

Path director cue and speed control cue are then used in normal approach made fashion. The change in function of the cue from path director to flight path is for a short time only. The use of a cue with a different form was considered but not deemed necessary since the control action with each is the same, i.e., fly to superimpose the image over the aimpoint.

The simultaneous use of a flight path cue and a path director cue was considered but eliminated because of resulting display clutter and the satisfactory control adequacy of use of the path director cue alone.

e. System 2 Alternate Displays

The following paragraphs list additional information which may be included in the System 2 display.

(1) Radar Altitude

The use of radar altitude (see figure 3-3.14) provides:

● Quantitative altitude information (horizon cue read against altitude scale)

● Qualitative altitude information (zero altitude reference read against horizon)

● Assessment of altitude situation

NOTE: The altitude scale moves at right angles to the horizon as a function of altitude. The altitude scale also moves directly with the horizon for pitch and roll attitude changes. If the spacecraft attitude is such that the horizon is not in the FOV, the altitude scale will also be out of view or may not be readable. In this case a numerical symbolic image is used at the right center edge of the display for altitude information.

(2) Selected Ground Track

The use of a cue for the selected ground track (see figure 3-3.14) provides information for:

● More rapid identification of the real world aimpoint and touchdown area

● Computer monitoring since the ground track cue (which is positioned by the computer) should be directly superimposed over the real world aimpoint and touchdown area

● More graphic indication of relative heading of spacecraft to the selected track, and lateral displacements from the selected track.

(3) Touchdown Area Cue and Aimpoint Cue

The use of a touchdown area cue and aimpoint cue (see figure 3-3.15) provides:

- Reference aimpoint and landing area cues for IFR condition for both assessment and control tasks. This may be of extreme importance if dust clouds obscure the real world vision and "instrument" operation is required

- Exact rapid identification of the real world aimpoint and touchdown areas

- Computer monitoring since these cues should be directly superimposed over the real world points.

The touchdown area cue may not be warranted for the general cases of approach and hover. Two specific cases where it may be of greater importance are:

a) Where the specific touchdown point has been preselected and it is desirable to indicate its exact location to the pilot.

b) Where there is considerable difference in position between the aimpoint and the touchdown area, such as would be the case if it were desirable or necessary to initiate the letdown at some altitude considerably higher than 50 feet, i.e., perhaps several hundred feet.

f. System 3 Display for Approach and Letdown

The System 3 display (see figure 3-3.16) is identical to that of System 2 with the addition of a perspective touchdown area cue and/or a max safe range cue.

The perspective touchdown area cue enhances the ability of the pilot to:

- Estimate altitude

- Estimate ground speed along course and cross course velocities as well as furnishing more effective displacement cues

- Estimate distances

- Estimate vehicle attitude

The max safe range cue enhances the ability of the pilot to:

- Insure that the selected aimpoint and touchdown area are within safe operational limits

- Select new aimpoint and touchdown area within safe operation limits

The use of the fuel remaining cue is optional since the size and shape of the max safe range cue is an indication of operational capability including the fuel remaining factor.

Assessment and control is the same as with the System 2 display but with the additional assessment information derived from the perspective touchdown area.

5. Advantages of the Projected Display System for Lunar Landing Operations

The use of the projected display system for lunar landing:

- Provides a heads-up display in flight modes where same is critically needed

- Provides a display of assessment and control information in the pilot's forward external field of view to improve visual flight control

- Presents synthetic visual information for use under "instrument flight" conditions such as may occur if dust clouds obscure the external view during final letdown

- Allows common display/control technique for both "visual flight" and "instrument flight" conditions

- Merges the pertinent instrument display information with the external view on a real 1:1 scale basis and with sensing compatible to real world sensing.

The projected display system specifically permits the following, while the pilot observes the normal external view:

- Rapid assessment of attitude by a 1:1 attitude scale and with real world sensing

- Simple determination of lateral and vertical position displacement from the selected descent path by the deviation ellipse cue relative to the real world aimpoint

- Simple assessment of present flight path angle relative to moon surface and simple automatic selection of a reference descent path to a real world aimpoint

- Clear simple indication of spacecraft orientation relative to the selected ground track

- Simple assessment of fuel remaining and its operational significance (max safe range cue)

- Simple path control through use of path control director cue (which includes any necessary "quickening" information) operated relative to the aimpoint (either real world or cue). The aimpoint is a natural center of interest point and thus inclusion of control information at this point is appropriate

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- Simple speed control through use of the speed control director cue (which includes any necessary "quickening"). The cue is appropriately located with the path director cue to center all control information at the natural center of interest point (i. e. the aimpoint)

- Simple altitude rate control during letdown by use of the vertical velocity director cue operated relative to the aimpoint.

- Rapid assessment of altitude situation by the altitude cue

- Rapid location and identification of real world aimpoint and touchdown area by projected cues for same

- Simpler assessment and estimation of altitude, along-course and cross-course ground velocities, distances and vehicle attitude by use of a perspective touchdown area cue.

ATT DIR

THRUST DIR

BORESIGHT REF

INITIAL MODE

REAL WORLD TARGET

ATT DIR

THRUST DIR
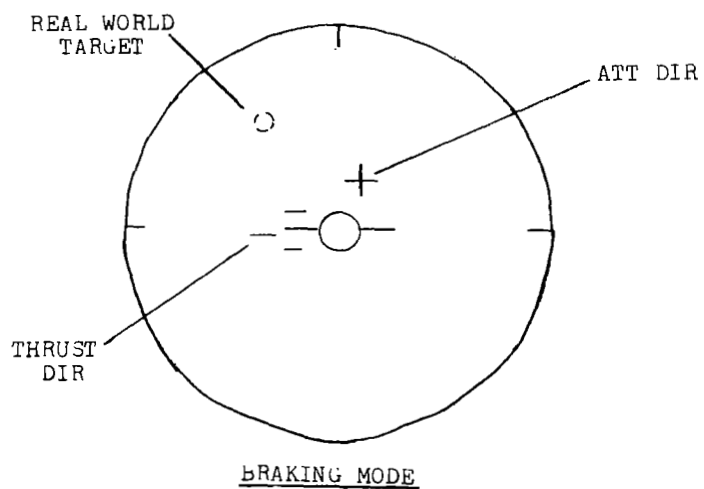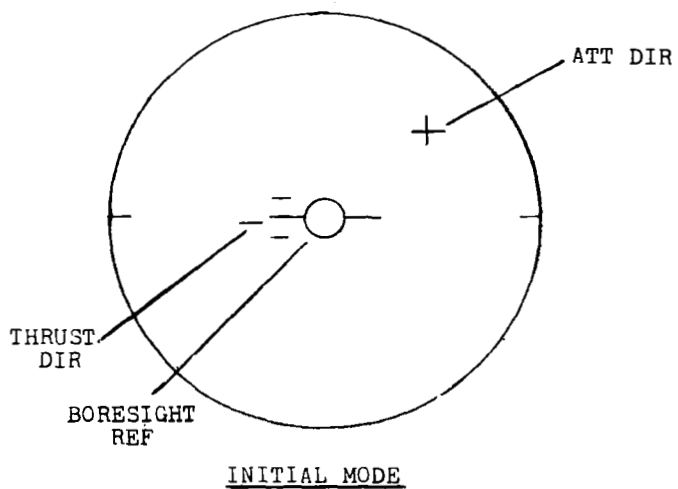
BRAKING MODE

FIGURE 3-3.1   DISPLAYS FOR RENDEZVOUS –
INITIAL MODE AND BRAKING MODE, SYSTEM NO. I DISPLAY

FIGURE 3-3.2  DISPLAYS FOR RENDEZVOUS—
INITIAL MODE, SYSTEM NO. 2 DISPLAY

RADAR
TGT

TGT LOS
RATE

PITCH-YAW
ATT DIR

REF INERTIAL
TGT

THRUST
DIR

BORE SIGHT
REF

SPACECRAFT ATTITUDE SO
TGT BRG WITHIN FOV
3-3.2A

PITCH-YAW
ATT DIR

RADAR
TGT BRG

PITCH-YAW
ATT REF

THRUST
DIR

BORE SIGHT
REF

(2)
REF INERTIAL
PLANE

(1)
TGT LOS
PERP. PLANE

SPACECRAFT ATTITUDE SO
TGT LOS PERP PLANE BRG
WITHIN FOV
2-2.2B

RADAR
TEST BRG

PITCH-YAW
ATT REF

THRUST
DIR

BORE
SIGHT
REF

SPACECRAFT ATTITUDE SO
TGT BRG & TGT LOS PERP PLANE BRG
NOT WITHIN FOV
3-3.2C

FIGURE 3-3.3  DISPLAYS FOR RENDEZVOUS —
INITIAL MODE, SYSTEM NO. 2 DISPLAY

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FIGURE 3-3.4 DISPLAYS FOR RENDEZVOUS—
INITIAL MODE WITH LOS RATE ARRESTED, SYSTEM NO. 2 DISPLAY

REF INERTIAL
TGT

RADAR
TGT

ATT DIR
(DEACTIVATED)

THRUST
DIR

SPACECRAFT ATTITUDE SO
TGT BRG WITHIN FOV
3-3.4A

ATT DIR
(DEACTIVATED)

RADAR
TGT BRG

ATT REF
(DEACTIVATED)

THRUST
DIR

TGT LOS
PERP PLANE

NEW REF
INERTIAL PLANE

SPACECRAFT ATTITUDE SO
TGT LOS PERP PLANE BRG
3-3.4B

ATT DIR
(DEACTIVATED)

RADAR
TGT BRG

THRUST
DIR

SPACECRAFT ATTITUDE SO
TGT BRG & TGT LOS PERP PLANE
BRG NOT WITHIN FOV
3-3.4C
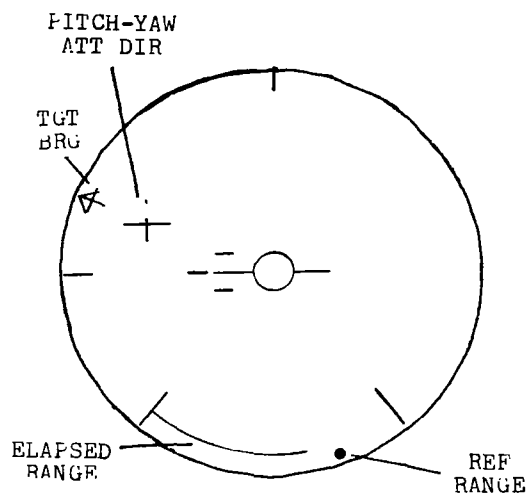
SPACECRAFT ATTITUDE SO          SPACECRAFT ATTITUDE SO
   TGT BRG IN FOV                  TGT BRG NOT IN FOV
      3-3.5A                          3-3.5B

FIGURE 3-3.5  DISPLAYS FOR RENDEZVOUS –
BRAKING MODE, SYSTEM NO.2 DISPLAY

TARGET IN FOV.
ATT REF NOT IN FOV.
3-3.6A

TARGET NOT IN FOV.
ATT REF IN FOV.
3-3.6B
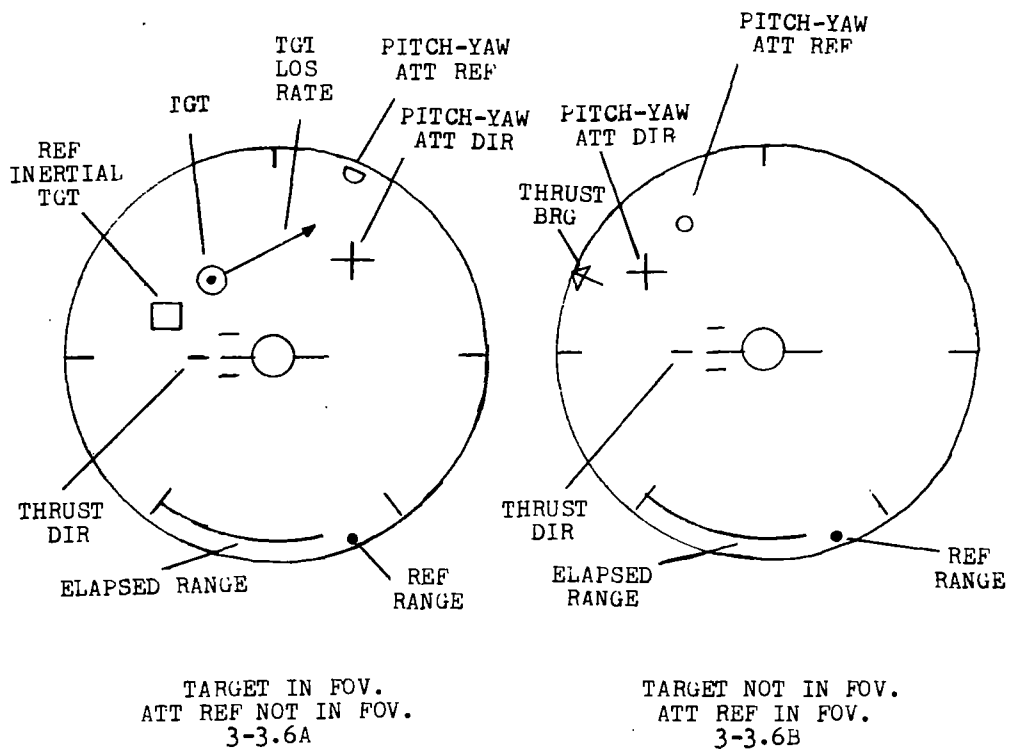
FIGURE 3-3.6 DISPLAY FOR RENDEZVOUS —
COMBINED INITIAL LOS RATE ARRESTMENT AND BRAKING MODE),
SYSTEM NO. 2 DISPLAY
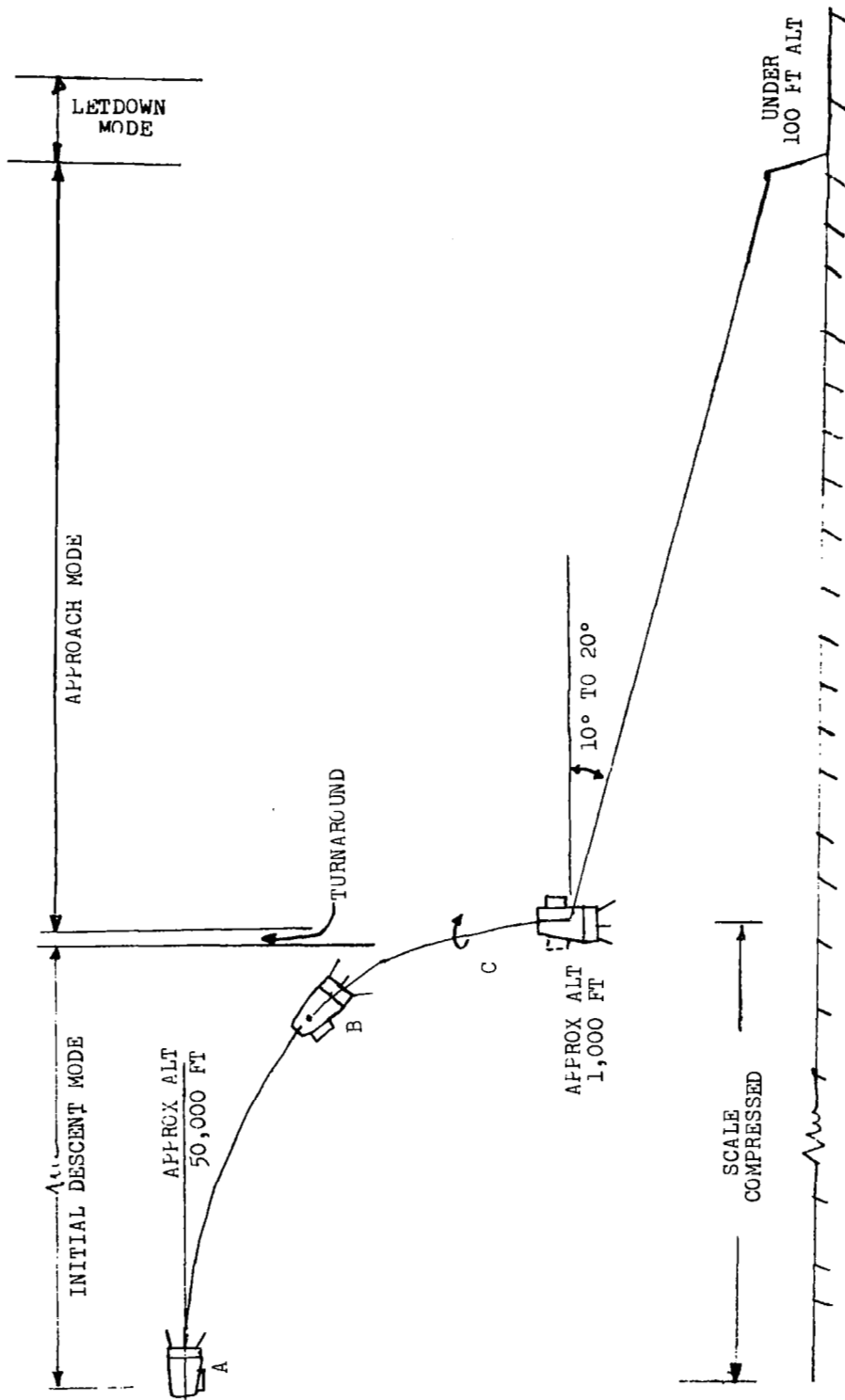
FIGURE 3-3.7 FLIGHT PROFILE – LUNAR LANDING
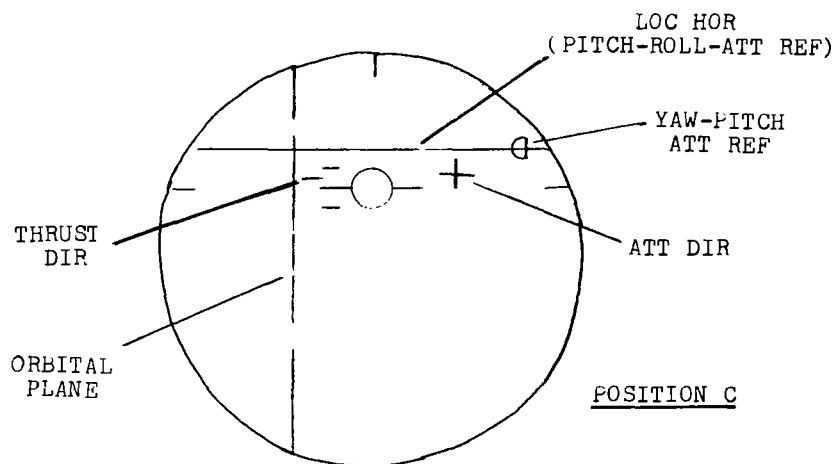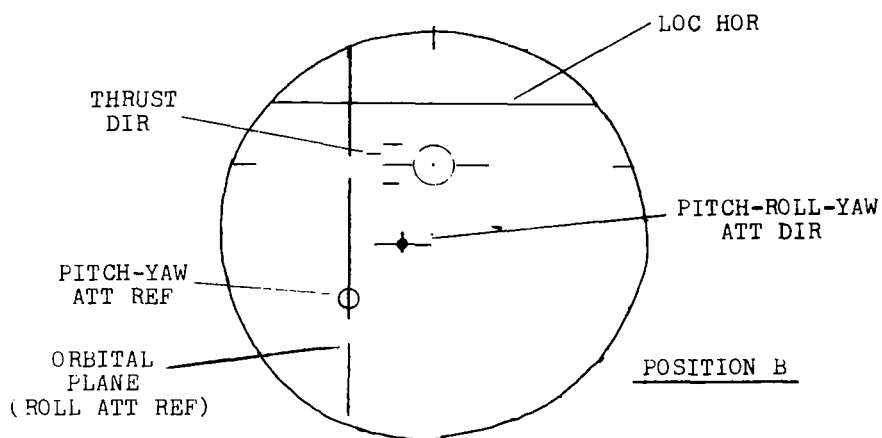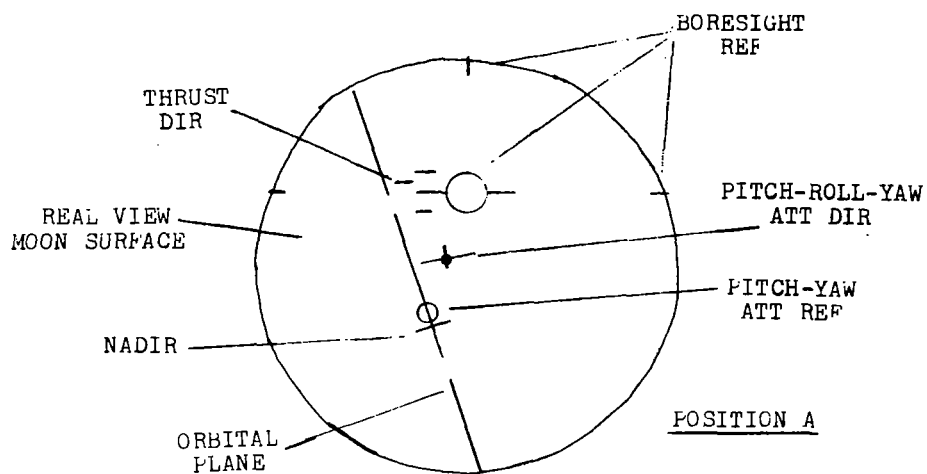
FIGURE 3-3.8  INITIAL DESCENT MODE DISPLAYS
FOR LUNAR LANDING, SYSTEM NO. 2 DISPLAY

THRUST
DIR

PITCH-ROLL
ATT DIR

ORBITAL PLANE
REF

VEL VECTOR
PERP PLANE
(PITCH-ROLL ATT REF)

POSITION A

YAW ATT REF

LOC HOR

THRUST
DIR

PITCH-ROLL
ATT DIR

ORBITAL PLANE
REF

VEL VECTOR
PERP PLANE
(PITCH-ROLL ATT REF)

POSITION B

YAW ATT REF

ORBITAL PLANE
REF

VEL VECTOR
PERP PLANE

YAW ATT REF
(FOR 180° TURNAROUND)

LOC HOR

THRUST
DIR

ATT DIR
(PITCH-ROLL-YAW)

POSITION C
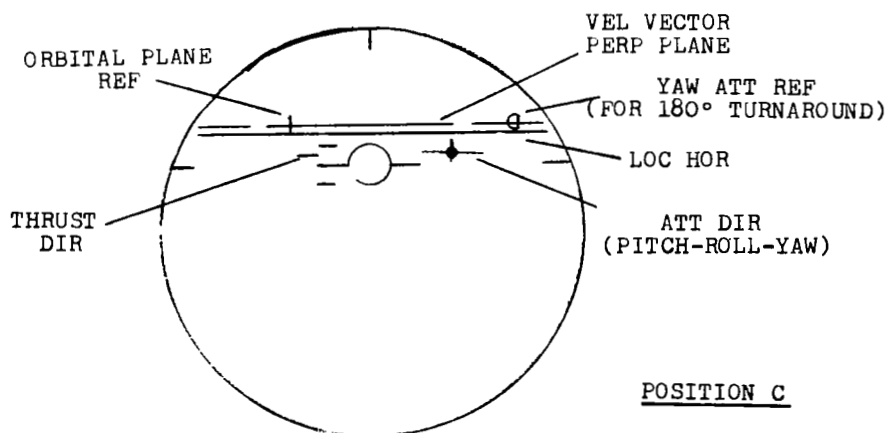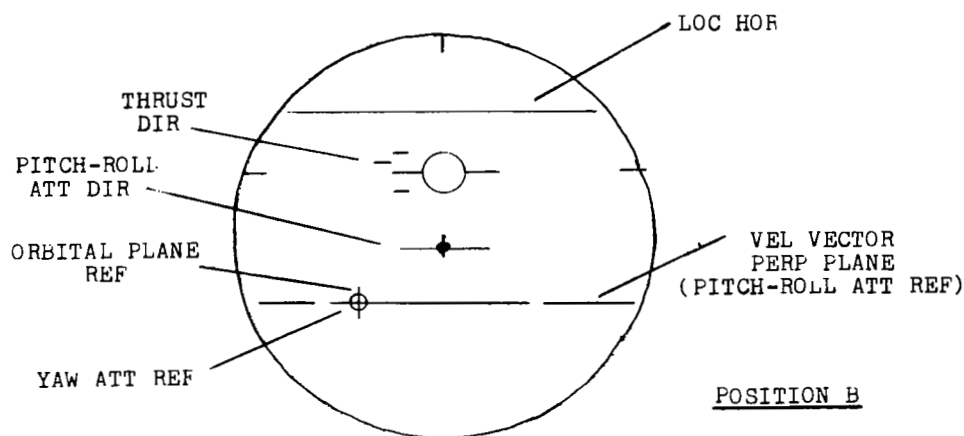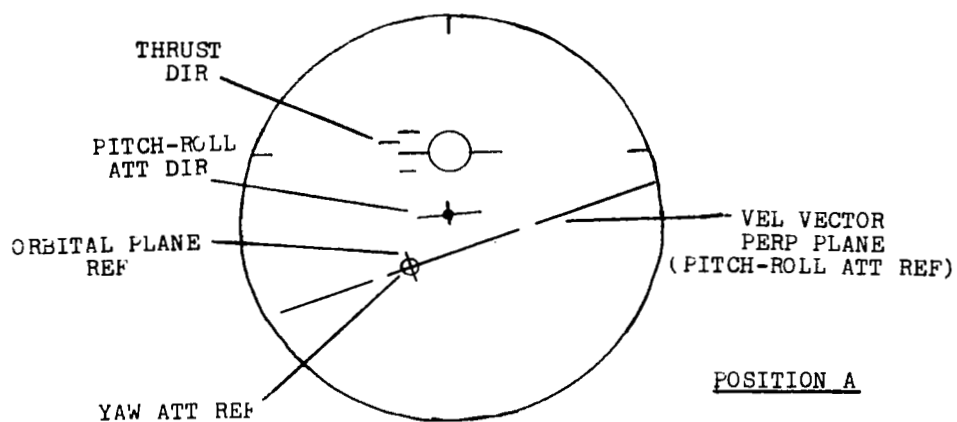
FIGURE 3-3.9    INITIAL DESCENT MODE ALTERNATE DISPLAY -
LUNAR LANDING, SYSTEM NO. 2 DISPLAY

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SIDE VIEW



PLAN VIEW
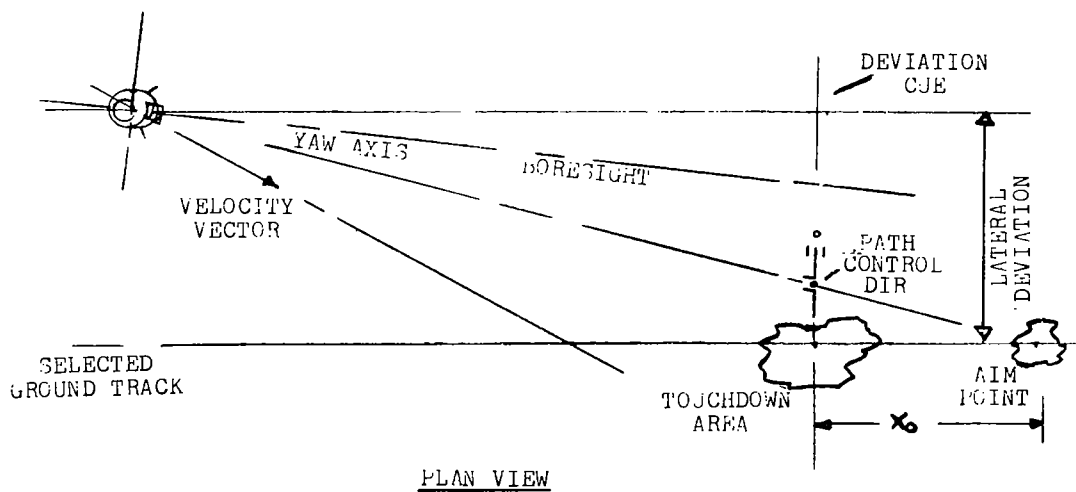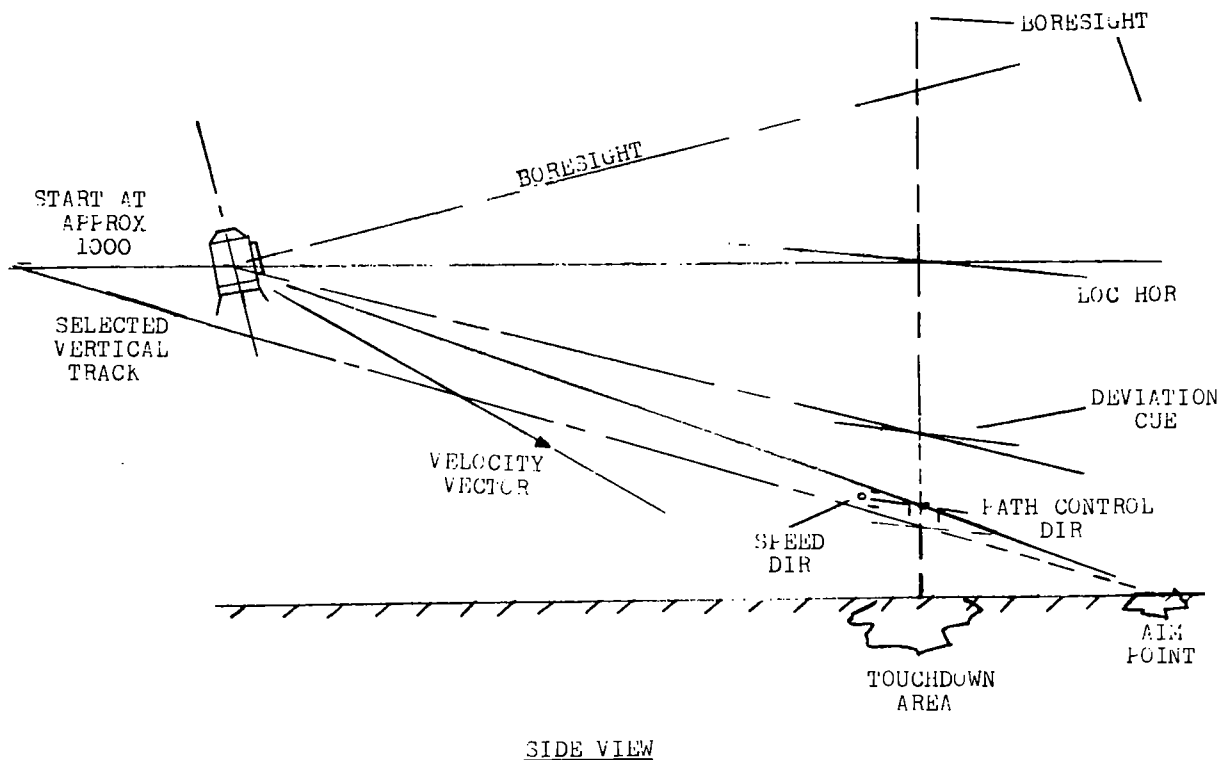
FIGURE 3-3.10. GEOMETRY
OF APPROACH MODE

FIG 3-3.11A



FIG 3-3.11B



FIG 3-3.11C
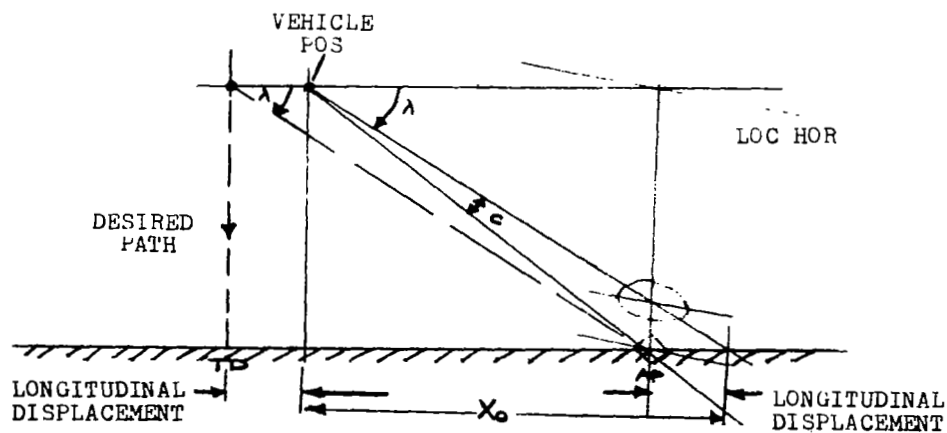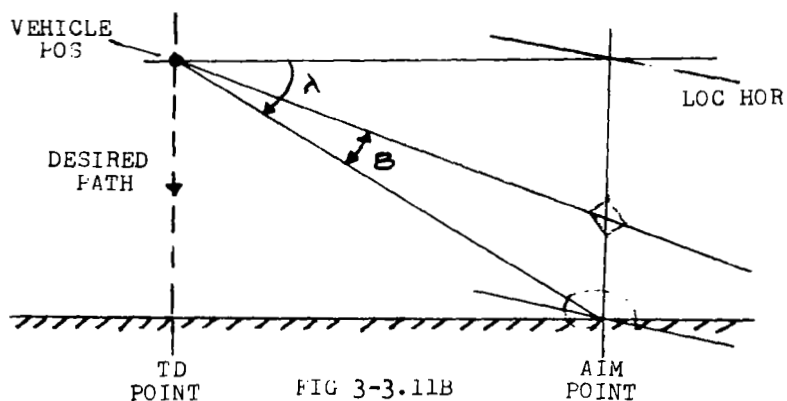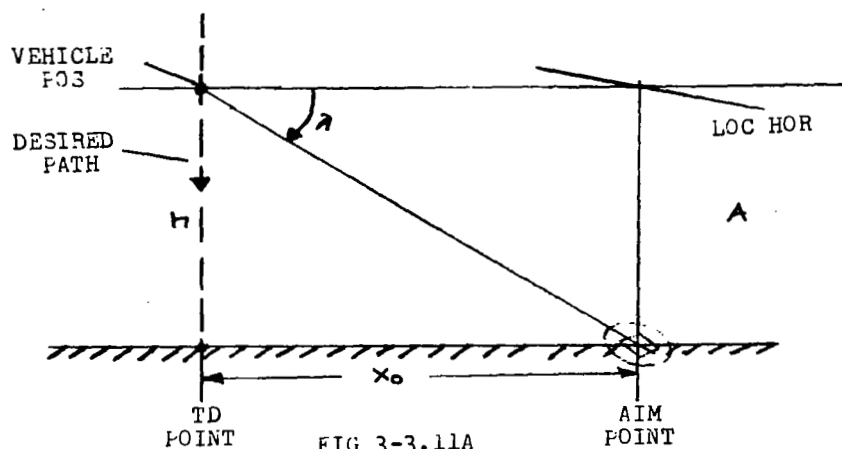
FIGURE 3-3.11. GEOMETRY OF
LETDOWN MODE (SIDE VIEW)

A.   APPROACH MODE

B.   LETDOWN MODE
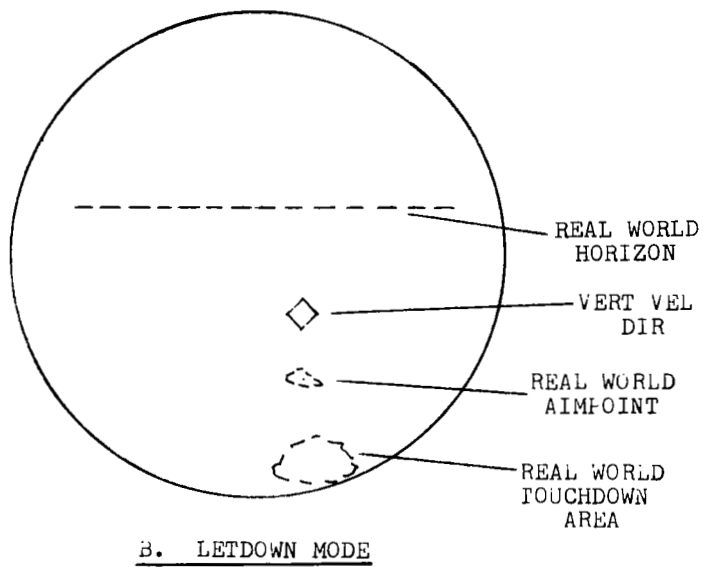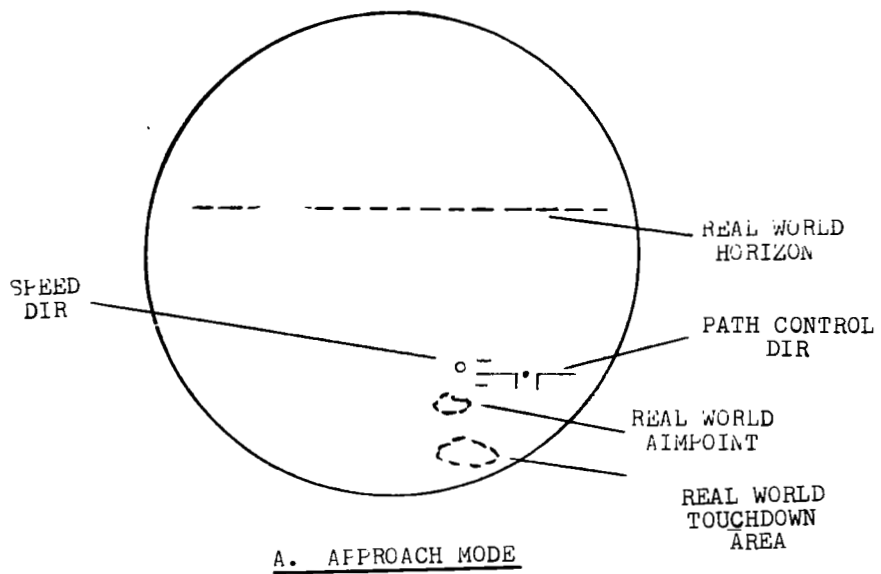
FIGURE 3-3.12. DISPLAYS FOR LUNAR
LANDING, SYSTEM NO. I DISPLAY

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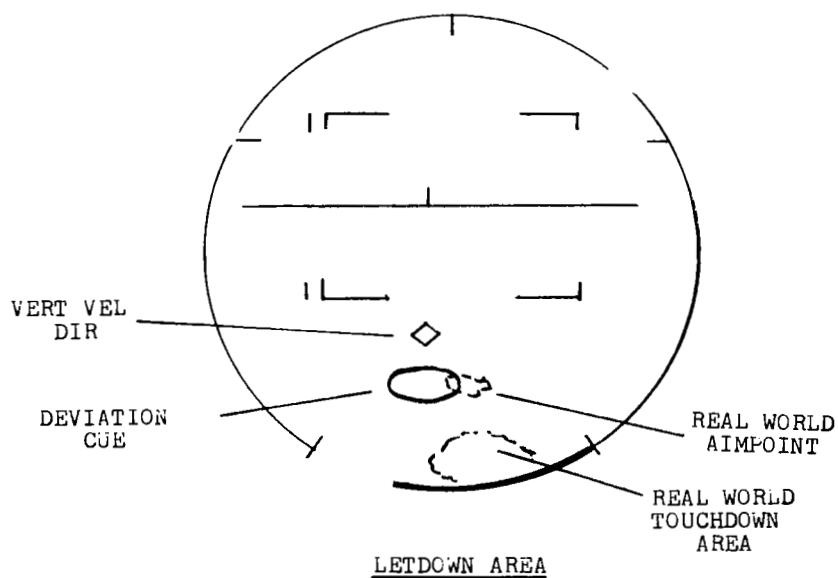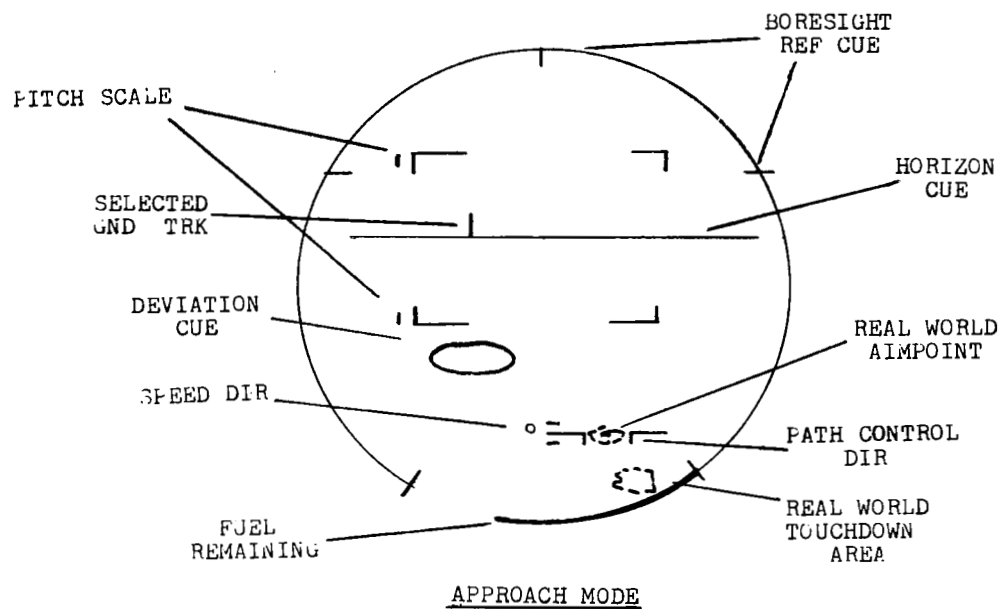
FIGURE 3-3.13. DISPLAYS FOR LUNAR
LANDING, SYSTEM NO. 2 DISPLAY
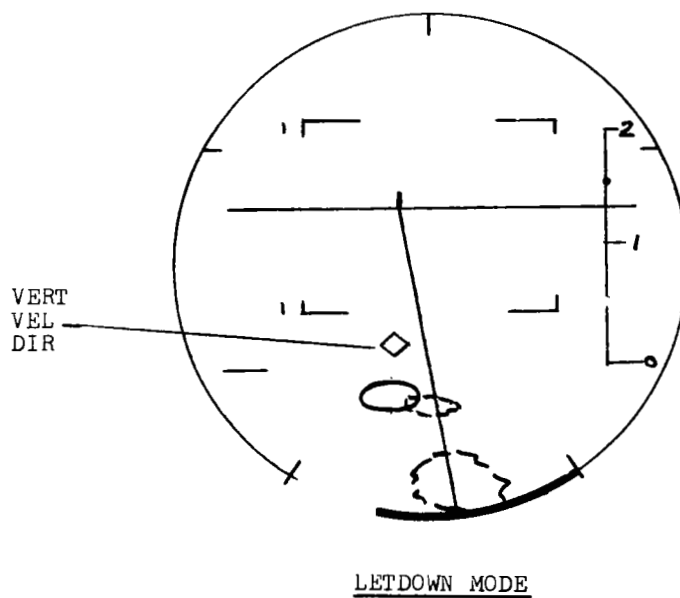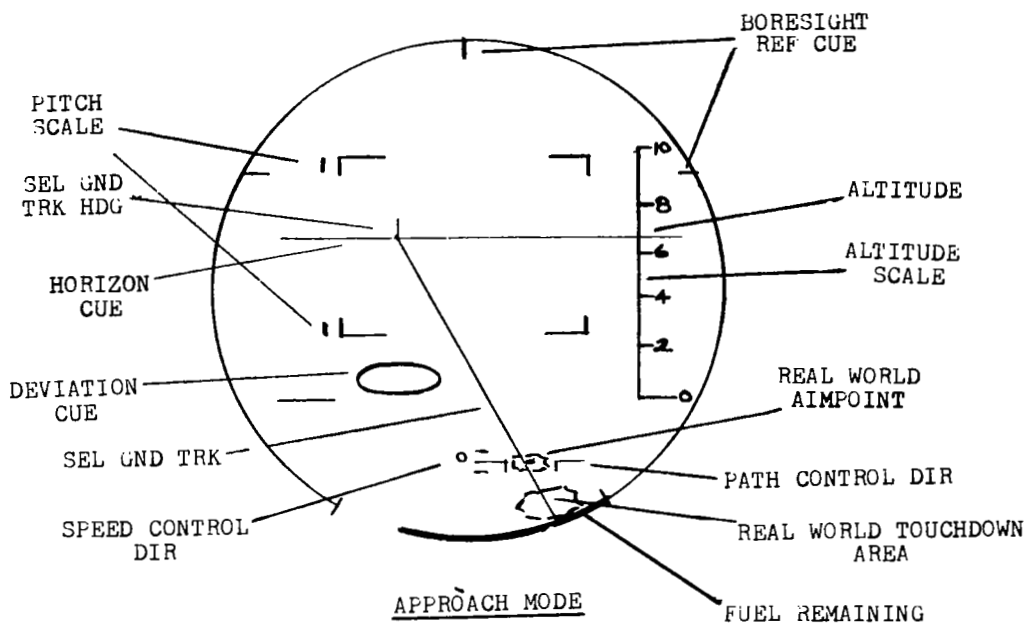
FIGURE 3-3.14.  DISPLAYS FOR LUNAR
LANDING, SYSTEM NO. 2 DISPLAY
(WITH SEL GND TRK AND ALTITUDE INFO)
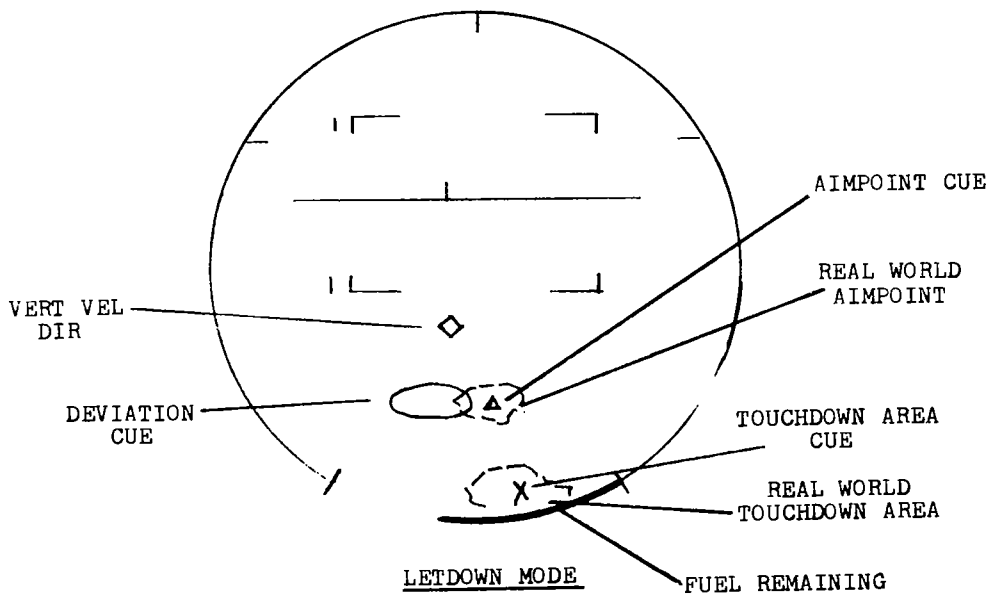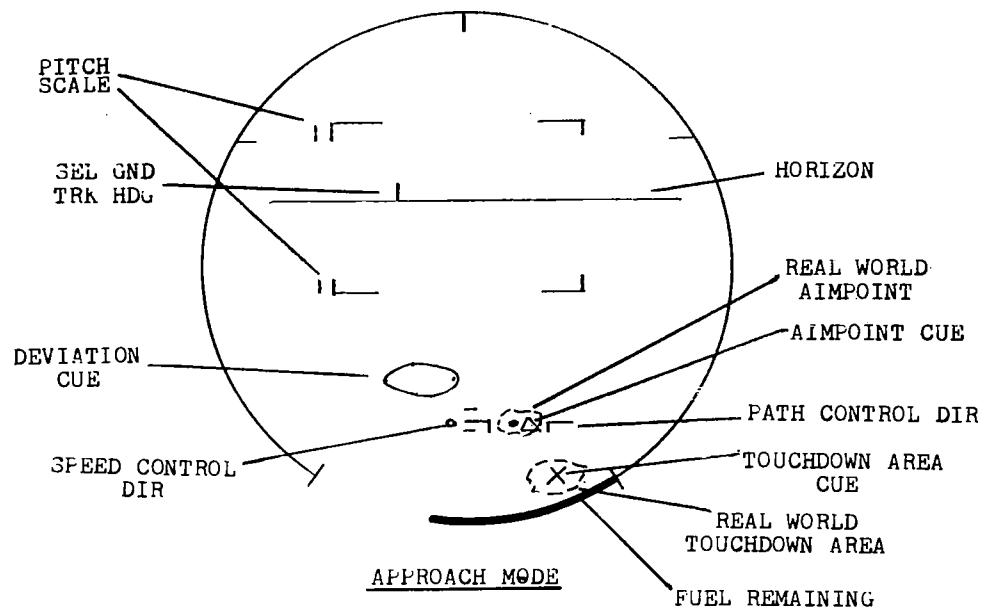
FIGURE 3-3.15. DISPLAYS FOR LUNAR
LANDING, SYSTEM NO. 2 DISPLAY
(WITH AIMPOINT AND T.D. INFO)
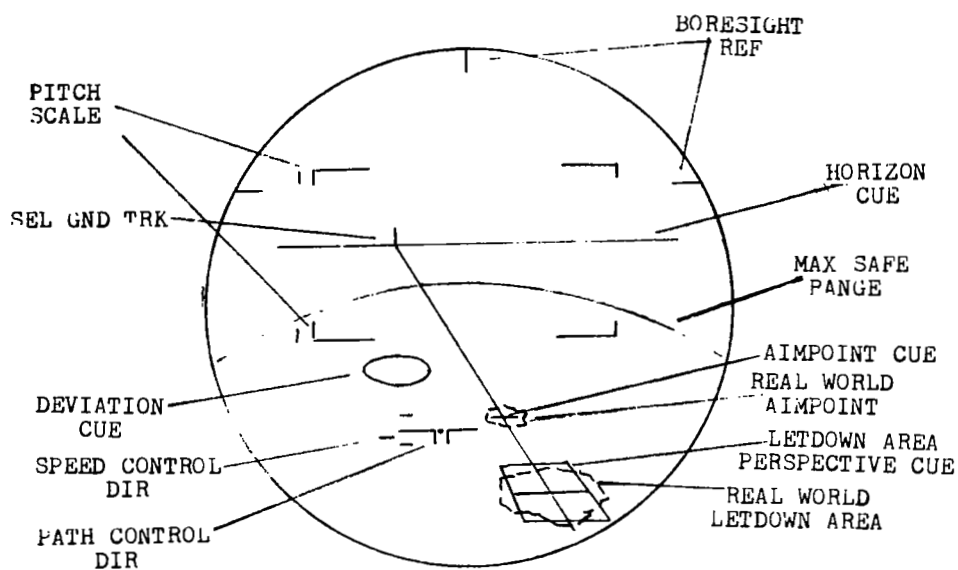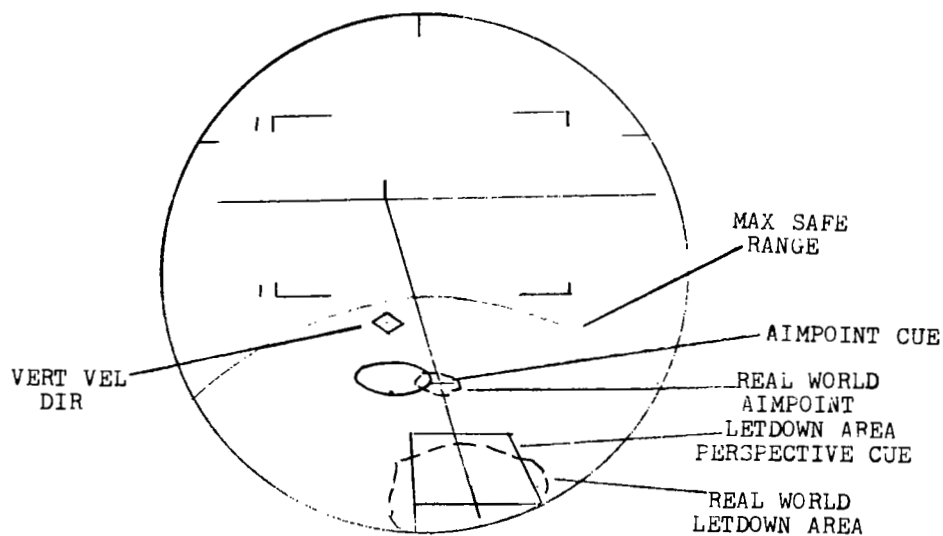
FIGURE 3-3.16. DISPLAYS FOR LUNAR
LANDING, SYSTEM NO. 3 DISPLAY
(PERSPECTIVE CUES)

## 3-4. PROJECTED DISPLAY/PANEL INTEGRATION ASPECTS

### A. ALTERED PANEL DISPLAY REQUIREMENTS

Of the three levels of projected displays, only levels 2 and 3 introduce the possibility of altering the information displayed on the panel. Level 1, which supplies only control with respect to a visually perceived target or landing site, has no equivalent in the panel. It is not until situation information such as target location or landing site location is introduced into the display that comparable situations exist in the projected and panel displays. With these levels of display complexity the opportunity exists to use the projected displays for purposes of indicating high resolution or fine grain data and eliminating this from the panel. In this case the panel can be used for purposes of gross or overall information, thus eliminating the troublesome requirement for presenting varying scales in order to cover the full range of data sensitivity that is required. An example is the case of altitude. On approach to to lunar landing a gross display indicating tens of thousands of feet is required at the beginning of the descent trajectory whereas at final touchdown a display resolution of a few feet is required. Similar situations exist with respect to range rate and velocity. Another area where the projected display may relieve a difficult panel presentation problem is in the area of basic orbital information. Here again is a situation in which the wide range of data sensitivity is required. For gross orientation the display system should indicate sufficient information to allow the astronaut to visualize his special problem in order to make use of his judgment and management capabilities. On the other hand, the minor corrections to the orbits for purposes of transfer or trajectory modification must be indicated with high display gain. The projected display offers the potential of presenting the fine grain orbital data with respect to the perceived outside view. The high resolution afforded by the optical technique is admirably suited for this purpose.

One of the most difficult requirements to satisfy in spacecraft instrument panels is the provision of lighting which will permit the instruments to be seen in any ambient light condition. A particularly troublesome problem exists for spacecraft because of the intense contrast between shadows and sunlit space within the interior of the craft. Conditions could occur during landing or rendezvous operations in which pilots might experience substantial interference due to difficulty of reading instruments, some of which are in shadow and others brightly lit by light streaming through windows. A projected display avoids this problem completely from the standpoint of not requiring eye adaptation from outside the display. The incorporation of a projected display would therefore logically reduce the lighting requirements for panel instruments with consequent savings of power and heat dissipation equipment.

## B. PANEL INSTRUMENTATION LOCATION

In a display complex which includes both projected and panel displays, the panel instrumentation location is generally substantially less critical than where the display configuration is totally on the panel. Heretofor, close proximity of panel displays to the window in spacecraft and aircraft has been the result of a requirement to minimize the visual effort and time involved in shifting the visual scan from instruments to the outside visual scene. The combined control and assessment tasks which demand integration of both visual data and instrument data leads to a requirement for this close proximity. A classic example in present day carrier aviation is the requirement to place the angle of attack apexer on the glare shield so that only slight shifts of the pilot's sight direction are required to obtain basic aerodynamic data during the landing approach to the carrier. The projected display has the feature, in control situations of this sort, of superimposing control and situation information directly and compatibly in the visual field. Hence the requirement for careful and close proximity of instruments to the window is considerably lessened. One aspect of this relaxation is to provide the option of presenting digital data in proximity to the window which otherwise would be forced to less favorable positions. An example might be the digital presentation of altitude and velocity immediately below the window in locations which otherwise would be absorbed by basic flight instrumentation.

## C. POSSIBLE DISPLAY CONFLICTS

In the case of the first level projected display, no real possibility of conflict with panel instrumentation exists. This is due to the fact that the display in used against visually perceived terrain. This is a situation which has no parallel in the panel. In the case of levels 2 and 3 where situation data is presented, conflicts will arise in the basic situation displays on the panel if a departure from the natural "inside-out" convention is used. Also the projected displays, by virtue of operating against visually perceived targets in both the rendezvous and landing situations, establish a sensing convention which is a "fly-to" the target. This particular sensing is frequently regarded as less desirable for the control of pointers in conventional panel instrumentation. In these cases the projected display should determine the standard of control sensing to be employed in the panel.

## 3-5. DISPLAY/CONTROL COMPATIBILITY

As a class of displays, the projected displays present no novel problems with respect to compatibility with the particular flight control techniques to be employed in spacecraft. The only distinction of consideration between the projected display and panel is the high resolution of the data which can be presented. This resolution, however, is equivalent to that obtained by the astronaut during direct visual reference to either rendezvous target or landing site, and hence compatibility with flight control is actually the same problem that exists with direct visual control of the vehicle. Suitable compatibility must exist, of course, when "director" or "quickened" types of control displays are being used. This compatibility is a requirement for all three levels of projected display. Here, the problem is to insure that the overall system response characteristics are satisfactory. This involves the display computation, human response to the demands of the display, and the vehicle response to the attitude and velocity control systems. This is essentially the question of the "flyability" of the display. This problem is usually most severe when the control power of the vehicle is marginal from a standpoint of response in relation to the demands of the maneuver being attempted. "Director" or "quickened" displays are best applied to critical operations of this sort, providing the display is capable of introducing a sufficient lead in the control display to offset the response deficiencies of the vehicle itself.

Quickening becomes more important in the control of the vehicle as the number of integrations between the astronaut's input and the variable being controlled (e.g., position, attitude) increases. For example, a control system in which attitude jets are employed to provide translation to a hovering vehicle represents a fourth order system. The astronaut's control input produces an angular acceleration which, after four successive integrations, results in a translational displacement of the vehicle. The human operator's task may be simplified by means of the quickening technique. The astronaut is continually presented with command information which is computed on the basis of the present state of the vehicle in relation to the desired end state, and he can satisfy these commands rapidly, with certainty. Higher order systems need not be fully quickened, however. The human operator has an effective control capability for first order systems, and second order systems to a more limited extent. He can exercise this control with a great degree of flexibility. Quickening may be employed to reduce the order of the system to a manageable level, while permitting the astronaut to retain the flexibility of manual operation. In the example of the hovering vehicle, the control of attitude may be quickened while the remaining translational functions are performed manually. Alternatively, attitude control may be manual while the translational control task is

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quickened. Some portions of the control loop may be made automatic. The exact control scheme will, of course, be dependent on the specific control task involved. In any event, the projection displays can always be made compatible with the control configuration.

In general, the projected display seems to present no unusual requirements in respect to the flight control characteristics of the vehicle during the orbital flight and rendezvous. Here the characteristics of the deliberate and preplanned actions or operations occurring at low relative speeds do not seem to be subject to difficult control problems. On the other hand, the lunar landing represents a situation in which close matching of the control display to flight control characteristics is mandatory if the full value of the projected type of display is to be achieved. Here the characteristics of the problem dictate a rapid and precise control of the landing in order to minimize the fuel requirements. This a typical case where the control power available is being used to the fullest. In Appendix C the results of a lunar landing control investigation have been presented, and the nature of the simultaneous attitude and lift engine control problems have been defined. The lunar landing is a particular operation in which the proper implementation of visual control displays should add significantly to the speed and precision with which an astronaut can accomplish site selection for final approach and landing. Full advantage may be taken of the visual inputs which can be afforded the pilot, as well as instrumentation which will supplement the basic deficiencies to be found in visual operations.

## 3-6. OPTICAL SYSTEM TRADE-OFF STUDY

### A. INTRODUCTION

The problem of optimizing the design of the optical system is one of the critical areas in heads-up display technology. The relationships of optical elements for given application parameters, such as field of view, viewing distance, and freedom of head motion, must be tailored to the particular vehicle cockpit space envelope. The resulting trade-off is time consuming unless generalized design and application charts are provided. A number of such trade-off analyses have been made and the results are presented in generalized form in this paragraph. An analysis of both the direct and relay optical systems has been conducted and both small angle gaussian optics and large field of view treatments are included.

Many of the symbols used in the analysis are self-evident from the illustrations; the more general symbols are defined as follows:

$D_x$ - Exit pupil diameter

$D_R$ - Entrance pupil diameter (relay lens aperture)

$D_c$ - Collimator diameter

$D_i$ - Image diameter (relayed)

$F_c$ - Collimator focal length

$F_R$ - Relay element focal length

$S_x$ - Entrance pupil (or relay lens position) to collimating element distance

$S_x'$ - Collimating element to exit pupil distance

$\beta$ - Half the image field of view angle

$B$ - Half the object field of view angle

$S$ - CRT (or reticle) to relay lens distance

$S'$ - Relay lens to relayed image distance

$D_o$ - CRT or effective reticle diameter

$u_o$ - Relay lens objective cone half-angle

## B. DESCRIPTION OF OPTICAL SYSTEMS CONSIDERED

### 1. Direct System Description

The direct system enables an observer to view the information from a CRT at the same time he is looking at the outside world. The CRT is placed at the primary focal plane of either a reflector or refractor so that the image is collimated and therefore appears to originate at infinity. There are three basic types of direct systems, spherical combiner, spherical mirror, and refractor. These systems are illustrated below.
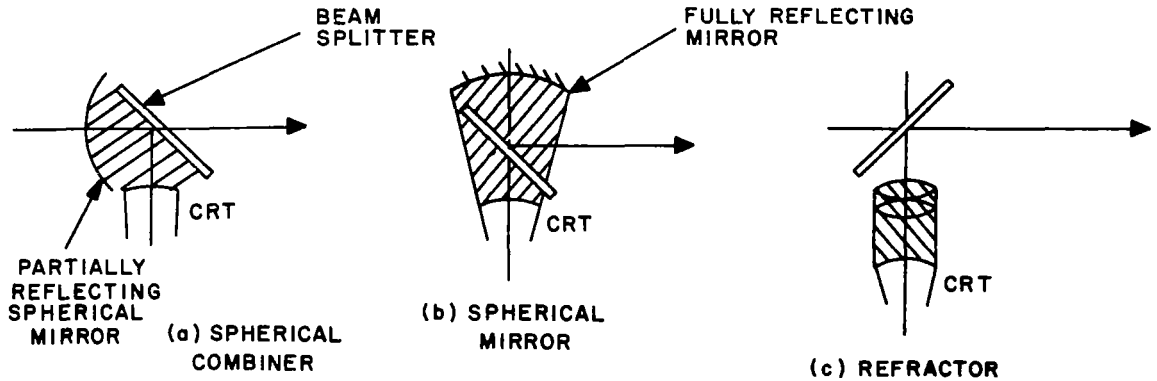


FIGURE 3-6.1. DIRECT OPTICAL SYSTEMS

The real world is seen through the partially reflecting spherical mirror in the spherical combiner system. The eye must be at the center of curvature in this case to avoid distortion of the real world. For the spherical mirror system, the real world is seen through only the plane beam splitter and the eye can be as close as a diameter away from the spherical mirror which is a fully reflective mirror. The refractive system similarly consists of the CRT to be viewed, located at the primary focal plane of the refracting lens with the real world viewed through the beam splitter.

The advantages of the direct system lie in its simplicity and in having the object and image remain near the optical axis. This eliminates the problems associated with off-axis aberrations, except when the field of view requirements are large.

### 2. Direct System Analysis

A collimating element is used to represent either the refractive lens or the reflective mirror with the object to be viewed located in the primary focal plane as illustrated in figure 3-6.2. Between the collimating element and the exit pupil, a beam splitter would be placed so that the outside real world may be viewed directly at the same time the desired information is superimposed in the same field of view. The beam splitter is

left out for clarity. Gaussian optic treatment is used in this analysis and therefore holds for small angles. Large angles are covered in a subsequent section.

A reasonable starting point is the consideration of the given system requirements and development of relationships which will serve as a basis for design. The primary requirements are instantaneous field of view $(2\beta)$ exit pupil diameter $(D_x)$ and viewing distance $(S_x')$. From figure 3-6.2 the required diameter of the collimator $(D_c)$ is determined in terms of the given parameters.

$$D_c = 2 S_x' \tan \beta + D_x \qquad (3\text{-}6.1)$$

In order for the projected object be in the image field of view, it must lie within the object field. The maximum object diameter $D_o$ which may be in this field is determined from the geometric relationships in figure 3-6.2.

$$\tan \beta = \frac{D_o}{2 (S_x - F_c)} = \frac{S_x' \tan \beta}{S_x} \qquad (3\text{-}6.2)$$

and since

$$S_x = \frac{F_c S_x'}{S_x' - F_c} \qquad (3\text{-}6.3)$$

substituting eq (3-6.3) in eq (3-6.2) results in:

$$D_o = 2 F_c \tan \beta \qquad (3\text{-}6.4)$$

$D_o$ is related to size and weight of the image generating equipment (CRT, etc.), and therefore, it is desirable to keep the quantity $D_o$ a minimum. If $\beta$ is fixed by the application requirement, then $D_o$ is directly related to $F_c$ from equation 3-6.4. However, there is a practical limit on how small $F_c$ may be, since an f/no. $(F_c/D_c)$ less than 1 is difficult to attain.

3. Internal Image Relay System Discription

The internal image relay system operates externally in the same manner as a direct system. The difference is in the use of the relay optics to magnify or to project the CRT image prior to its collimation. There are, however, the two following advantages that the relay system makes available to the system designer.

- Use of a smaller CRT for the same field of view.

- Projection of the CRT image to the collimator focal plane from a remote position.

The CRT is placed at an object position of the relay lens so that its image is formed at the primary focal plane of the collimating element, and therefore the virtual image seen by the observer seems to originate at infinity. This image is superimposed on the real world by one of the three basic types of relay systems: (1) the reflective spherical combiner relay, (2) the reflective spherical mirror relay, and (3) the refractive relay. Typical applications of these three relay systems are illustrated in figure 3-6. 3.

The eye should be placed at the center of curvature of the partially reflective combining mirror so that the real world appears undistorted. Since the relayed image of the CRT is at the primary focal plane of the mirror, the observer would see the image superimposed in the far distant field of the real world. One of the problems associated with this system is the distortion problem resulting from the off-axis astigmatism aberration. However, despite this problem there is a considerable saving in weight over the refractive relay system.

The fully reflective spherical mirror relay system operates in a manner similar to the previously described system and without the severe off-axis problem.

4. Internal Image Relay Display System Analysis

For simplicity, the collimating element, which in figure 3-6. 4 is represented as a lens, is used to represent either a reflector or a refractor. The beam-splitter is left out for clarity so that an on-line optical system may be analyzed. It is to be noted that the direct system was analyzed in a direction opposite to the rays passing through the system, since the input constraints are on the viewing end of the system. This point of view is again taken in analyzing the internal image relay system. The primary requirements are the same as the direct system and equation (3-6. 1) is used to determine the collimating element diameter. The diameter of the image $D_i$ is determined from equation (3-6. 5) based upon figure 3-6. 5 and the use of gaussian optics.

$$D_i = 2 F_c \tan \beta \qquad (3-6.5)$$

The diameter of the projection lens, $D_R$ which serves as the entrance pupil is directly determined from figure 3-6. 5.

$$D_r = \frac{D_x S_\lambda}{S_x'}$$

Since $D_x$ and $S_x'$ are set in the application requirements, the only variable in equation (3-6.6) is $S_x$. A minimum value for $S_x$ is desirable; however, the magnification function of the projection element constrains this value. It is seen from figure 3-6.6 that:

$$S_x = S' + F_c \qquad (3\text{-}6.7)$$

It is desirable to have a very short focal length ($F_R$) for the projection element. Again, a practical f/no. constrains the choice, since the diameter is determined by eq 3-6.6. Once a minimum workable $F_R$ is chosen, one may determine the object position (S) from the gaussian form of the thin lens formula.

$$\frac{1}{S} + \frac{1}{S'} = \frac{1}{F_R} \qquad (3\text{-}6.8)$$

The object diameter $D_0$ may now be determined from the magnification performed by the relay lens

$$M_R = \frac{D_i}{D_0} = \frac{S'}{S} \qquad (3\text{-}6.9)$$

so that the original field of view requirement is met. The position of the relay lens may be determined directly after the focal length of the collimator has been determined and the required viewing distance from the following relationship:

$$S_x = \frac{S_x' \, F_c}{S_x' - F_c}$$

Similarly, the position of the CRT may be determined directly from the focal length of the relay lens from

$$S = \frac{S' \, F_R}{S' - F_R}$$

## C. OPTICAL INVARIANT FOR WIDE ANGLE SYSTEMS

The previous analyses have derived optical system relationships for narrow field systems, in which the geometry is easily understood. The gaussian, narrow field approach is not applicable without error to wide-field systems. Therefore the use of the optical invariant to simplify the analysis

of complex wide-field optical systems, such as the internal image relay system discussed herein. Since the trade-off results are to be applicable to wide-field systems, the wide angle form of the invariant is used. The invariant is based upon Abbe's sine condition, which is a requirement for a coma free system. The invariant is stated as: $ny \sin \theta = n'y' \sin \theta'$

This optical invariant is applicable to an entire optical system and may be used between any two object and image spaces. Applying the invariant to the internal image relay display system, as illustrated in Appendix F, it takes the form:

$$\frac{D_0}{2} \sin u_0 = \frac{D_x}{2} \sin \beta \qquad (3\text{-}6.10)$$

since $\qquad n = n'$

or $\qquad \sin \beta = \dfrac{D_0 \sin u_0}{D_x}$

since $\qquad \sin u_0 = \dfrac{D_R}{2S} = \dfrac{1}{2f_0}$

therefore

$$\sin \beta = \frac{D_0}{2 \, D_x f_0} \qquad (3\text{-}6.11)$$

Equation (3-6.11) serves as the basis for the development of the relay system design nomograph (figure 3-6.9) together with the geometric relationships. The Smith-Helmholtz and the Gaussian form of the invariant are derived for small angles, and it may be observed that Abbe's sine condition reduces to the same relationship where the angles are small. (i.e., $\theta \approx \tan \theta \approx \sin \theta$).

$$ny \tan \theta = n'y' \tan \theta' \text{ Gaussian} \qquad (3\text{-}6.12)$$

$$ny \theta = n'y' \theta' \text{ Smith-Helmholtz} \qquad (3\text{-}6.13)$$

In comparing the results obtained using wide angle optics with the results using gaussian optics in determining the diameter of the collimator the difference lies only in the use of the sine and tangent function.

$$D_c = 2 S_x' \tan \beta + D_x \qquad \text{(gaussian optics)}$$

$$D_c = 2 S_x' \sin \beta + D_x \qquad \text{(wide angle optics)}$$

Again, the two results are in agreement for cases with small field of view angles.

The optical design and trade-off charts presented in this report all use the wide-angle relationships and are therefore not restricted in their use.

## D. LUMINANCE CONSIDERATIONS

The luminance of a projected collimated image is, except for transmission and reflection losses, the same as the luminance of the object, which is either a CRT or an illuminated reticle. That this luminance invariant exists may be seen from the following, where $\bar{y}_O$ = object height; $n_O$ = index of refraction in object space; $u_O$ = slope angle of rim ray; $\bar{y}_i$ = image height; $n_i$ = index of refraction in image space; and $u_i$ = slope angle of rim ray in image space.

$$\bar{y}_o \, n_o \, u_o = \bar{y}_i \, n_i \, u_i \qquad (3\text{-}6.14)$$

Equation (3-6.14) is the small angle optical invariant, as derived in Military Standardization Handbook MIL-HDBK-141, page 6-6, written for the object-image planes.

The following simplified treatment is applicable for rays making small angles with the optical axis (small field angle) for which the luminance is a maximum. Actual luminance falls off at field angles away from the optical axis. Squaring equation (3-6.14)

$$\bar{y}_o^2 \, n_o^2 \, u_o^2 = \bar{y}_i^2 \, n_i^2 \, u_i^2 \qquad (3\text{-}6.15)$$

Since $y_o^2 \propto$ the area of the object $\propto A_o$

$u_o^2 \propto$ solid angle subtended by entrance pupil of optical

system $\propto \Omega_o$ (for **small angles**)

$\bar{y}_i^2 \propto$ area of image $\propto A_i$

$u_i^2 \propto$ solid angle subtended by exit pupil of optical

system $\propto \Omega_i$ (for **small angles**)

$$n_o^2 \; A_o \; \Omega_o = n_i^2 \; A_i \; \Omega_i \qquad (3\text{-}6.16)$$

The light flux leaving the optical system equals the product of the flux entering the system and the transmittance of the system, T.

$$TN_o \; A_o \; \Omega_o = N_i \; A_i \; \Omega_i \qquad (3\text{-}6.17)$$

in which N = luminance (brightness)

Divide equation (3-6. 17) by equation (3-6. 16)

$$T \; \frac{N_o}{n_o^2} = \frac{N_i}{n_i^2} \qquad (3\text{-}6.18)$$

Equation (3-6.18) states the $N/n^2$ is an invariant in the direction of a ray as it goes from one medium to another. If $n_o = n_i$, then

$$T \; N_o = N_i, \qquad (3\text{-}6.19)$$

which is the luminance invariant, stating that the luminance N is invariant along a ray in an isotropic medium, except for absorption losses.

## 1. Typical Results

A projection relay lens has been designed which has 10 air-glass surfaces, which would require a corrector plate with two or more surfaces, making a total of 12. Each air-glass surface is coated with magnesium fluoride to reduce reflections to less than 2 percent. Without coating the loss per surface would be 5.3 percent, with an average index of refraction for the glass of 1.6. For <u>normal</u> incidence

$$R = \text{Reflectance} = \left(\frac{n-1}{n+1}\right)^2 = 0.053 \qquad (3-6.20)$$

Using a conservative figure for average transmittance of 0.98 per coated-glass surface, the following system transmittance would be found:

$$T = (0.98)^n, \qquad (3-6.21)$$

where n = the number of coated surfaces.

For a system with 12 air-to-glass surfaces (n=12), a coated lens system would have a transmittance of 0.784, and an uncoated system would have 0.518. The corresponding transmittances for 24 surfaces are 0.616 and 0.269.

## 2. CRT Luminance

Measurements made on the 3-inch CRT used in a Sperry display system show that the luminance of the tube is 7200 foot-lamberts. In order that the luminance of the collimated image be 900 foot-lamberts, the overall transmittance of the entire optical system must be at least 0.125. A reflectance at a semi-mirror of 0.16 together with a transmittance of 0.784 will produce a luminance of 900 foot-lamberts. With a reflectance of 0.2 at the combining mirror, a projection system with 23 coated air-glass surfaces could be used. Therefore, with the existing CRT, even the Frankford Arsenal overlapping monocular periscope could be used, which has 20 air-glass surfaces between the CRT and the observer.

## E. DESCRIPTION OF THE OPTICAL DESIGN CHARTS

### 1. Direct System

In determining the relationships between total field of view ($2\beta$), CRT diameter $D_0$, eye motion (H), at various viewing distances $S_x'$, one may

refer to subparagraph B where the equations are developed. However, in practical applications it is more convenient and less time consuming to make use of a design chart relating these variables. Figure 3-6.7 is such a chart developed for the f/1 and f/2 direct systems. These charts are applicable for viewing distances ($S_x'$) with the eye positioned at the center of curvature or the focal plane of the collimator. For a specified field of view ($2\beta$), the variation of collimator diameter with CRT diameter is shown, and the head motion required to utilize the field of view is also indicated. This chart is based on the wide angle form of equation 3-6.4, i.e.,

$$D_o = 2 F_c \sin \beta$$

for the CRT diameter ($D_O$)/collimator diameter ($D_c$) relationship. In the f/1 chart, $F_c = D_c$, and in the f/2 chart, $F_c = 2 D_c$, for the cases in which the CRT diameter furnishes an image field angle larger than the angle subtended by the collimator diameter for the specified viewing distances, i.e., at the focal plane ($S_x' = F_c$), or the center of curvature ($S_x' = 2 F_c$); the required eye movement is also indicated.

Figure 3-6.7 may be used to obtain the image field provided by any CRT diameter and collimator focal length. The f/1 portion of the figure is used, for which the collimator diameter and focal length are the same. The image field is obtained from the intersection of the CRT diameter and focal length lines (collimator diameter).

2. Direct and Relay System

A general chart (figure 3-6.8) which may be applied to both direct and relay systems has been developed. This chart relates field of view ($2\beta$), collimator diameter ($D_c$), viewing distance ($S_x'$), and eye motion (H), for both reflective and refractive optics. The field of view variation with collimator size, viewing distance, and eye motion is shown. CRT or reticle diameter required to generate a given field of view is not obtainable from this chart and is found in figure 3-6.7 for a direct system, or figure 3-6.9 for a relay system. Figure 3-6.8 is directly applicable to refractive and fully reflective mirror systems. The viewing distance is restricted to the center of curvature ( =2x focal length) for spherical combiner systems to avoid distortion of the real world.

3. Relay System

A nomograph has been developed (figure 3-6.9) to be used to establish the characteristics of a relay system. The derivation of the equations

for the nomograph is given in Appendix F. A brief description of its application is also included here. For a given CRT diameter, $(D_O)$, and a specified field of view, $2\beta$, scales 1 and 8 are used to determine the parameter Q. For the field of view, $2\beta$, exit pupil size, $D_X$, and viewing distance one may determine the collimator diameter either from the equations developed in subparagraph B or from figure 3-6.8. For a given f/no. the collimator focal length $F_C$ is directly determined so that the relay magnification $M_R$ may now be determined $(M_R = F_C/Q)$. In order to determine the f/no. of the relay lens $(f_R)$, proceed from the exit pupil diameter scale 4 through the Q originally determined to scale 7 $(f_O/no.)$. From this point on scale 1 proceed to the relay magnification $(M_R)$ scale 10 and where this line intersects scale 9, the f/no. of the relay lens is determined. This is a basic design factor since lens design complexity is inversely related to f/no. For this given configuration, the field of view available as a function of CRT diameter is obtained from scales 1 and 8 using the original value for Q on scale 5 as a pivot point.

## F. OPTICAL TRADE-OFF CHARTS

A group of nine optical system trade-off charts, figures 3-6.10 through 3-6.18, have been developed to show volume and weight for refractive and reflective systems for various field of view angles, viewing distances, and exit pupil sizes. Both direct and relay systems are included.

### 1. Analysis of Volume and Weight

The basis for the volumes and weights for direct systems may be described with the aid of figures 3-6.1, a, b, and c. The volume is computed as that of a section of cone of length $F_C$, with collimator and CRT diameters at each end for the reflective system. This is conservative for the spherical combiner system. The refractive system has an additional volume to account for the thickness of the lens cell. The weight of the reflective system consists of the weight of the spherical mirror, the beam splitter and a partial enclosure from CRT to collimator. The weight of the refractive system consists of the weight of the lens, beam splitter, and full enclosure. The analysis has been conducted for an f/1 collimator with exit pupil diameters of 0.5, 1.0, 3, and 5 inches. Viewing distances $(S_X')$ from 5 to 40 inches and total field of view $(2\beta)$ to 60 degrees have been covered.

The maximum fields of view in figures 3-6.10 through 3-6.13 for the direct systems for each viewing distance and exit pupil have been based on a practical minimum viewing distance limitation. It is assumed that a plane beam splitter is needed in all the direct systems. (This is certainly true for the fully reflective and the refractive cases and true for an on-axis spherical

combiner.) The minimum viewing distance is, therefore, on the order of one collimator diameter. Shorter distances would interfere with the beam splitter or block part of the collimator. The maximum field of view is thus given by the wide angle form of equation 3-6.1 for $S_x' = D_c$.

$$2\,\beta_{max} = 2\sin^{-1}\frac{(D_c - D_x)}{(2\,D_c)}$$

The above equation, a weight of 150 pounds, a volume of approximately 3000 in$^3$, or a CRT diameter of 10 inches were used as limiting points for the curves in figures 3-6.10 through 3-6.13.

For the f/1 refractive system the length of the collimator ($L_c$) is assumed to be linearly related to the collimator diameter ($D_c$) by

$$L_c = 1.4\,D_c$$

while for the f/2 system $L_c$ is assumed to be 0.5 $D_c$. The $L_c = f(D_c)$ relationship from which the factors for f/1 and f/2 have been obtained has been estimated on the basis of current optical designs for similar wide field, highly corrected systems.

The thickness of the mirror is assumed to be 0.25 inches for collimator diameters of less than 12 inches and 0.375 inches for diameters greater than 12 inches. The glasses employed in these low f/no. systems will consist of heavy flints as well as crowns and, therefore, an average density of 0.12 lbs/in.$^3$ is assumed for the glass itself. Based upon a known lens design of this type, the average density for lens cells is assumed to be 0.04 lbs/in.$^3$. The enclosure is assumed to be made of aluminum 1/8-inch thick with a density of 0.1 lbs/in.$^3$.

The volume and weight have been determined for the relay systems as illustrated in figures 3-6.3, a, b, c. The volume for the reflective relay system consists of the volume of the section of a cone from the CRT to relay lens and the section of the cone from the relay lens to the collimator. The volume of the refractive system also includes an allowance for the thickness of the collimating lens. The weight of the combining spherical reflector relay system includes the weight of the relay lens, spherical mirror and enclosure from the relay lens to the CRT. For a fully reflective relay system the weight of the beam splitter would be added, although not included in these charts. The weight of the refractive relay system includes the relay lens, collimator lens, beam splitter, and enclosure from CRT to relay lens, as well as the enclosure from the relay lens to the collimator.

The analyses have been conducted for two relay system combinations. The first includes an f/0.75 relay lens with an f/1 collimator, and the second an f/1.5 relay lens with an f/2 collimator. These two combinations represent current state-of-the-art (f/0.75, f/1), and a less complex optical design (f/1.5, f/2).

Viewing distances of 5 to 40 inches over a total field of view to approximately 40 degrees are included. In order to restrict the relay optical systems to practical configurations from an installation standpoint, an assumption was made that the relay lens aperture would be no greater distance from the collimator than twice the viewing distance. The basic optical relationships were then used to establish viewing distance limitations as functions of collimator f/no. field of view, and exit pupil size. These limitations are included in the relay system trade-off charts. The minimum viewing distance for these conditions is 1.5 x collimator focal length.

For the f/0.75 relay lens, the relay cell length ($L_R$) is assumed to be twice the lens diameter, and the f/1.5 relay lens length is assumed to be 0.75 x the lens diameter, using the same L = f (D) relationship previously discussed. Other assumptions and conditions are the same as those noted for the direct system.

## 2. Results of Volume and Weight Analysis

The volume, weight and CRT diameter ($D_O$) have been presented in figures 3-6.10 through 3-6.18 as a function of total field of view ($2\beta$) for viewing distances ($S_x'$) of 5, 10, 20 and 40 inches, for exit pupils of 0.5, 1, 3, and 5 inches for both the direct and relay refractive and reflective systems. The term exit pupil is used here as it is technically defined; therefore, the total field of view is observable from any point within the boundary of the exit pupil. The significance of the exit pupil is further discussed in subparagraph G below.

Figures 3-6.10 through 3-6.13 present the weight and volume trade-off charts for the direct system. Figures 3-6.14 through 3-6.17 present the weight and volume trade-off charts for the f/1:f/0.75 relay system. Figure 3-6.18 is included for the f/2:f/1.5 relay system for a 3 inch exit pupil to show the effect of f/no. on volume and weight.

The increased weight and volume for the f/2:f/1.5 system for the same viewing distance and field of view, as compared to the f/1:f/0.75 system, are due to larger relay lens, larger CRT, and increased optical system length which result from the larger focal lengths. The collimator is the same diameter in both cases, although of different focal lengths.

In general, there is an increase of volume, weight and CRT diameter with an increase of the field of view and exit pupil diameter. These parameters also have larger values as the viewing distance is increased. The reflective system results in considerable savings of weight; however, it must be kept in mind that an off-axis reflective system presents a problem in correcting off-axis image distortion. The relay lens sizes and weights are conservatively based on an off-axis design, however,

The fact that the curves for the relay system do not follow a fixed pattern for various viewing distances is a result of the complex interaction of parameters in a relay system.

## G. APPLICATION OF THE OPTICAL CHARTS

In normal use of an optical viewing system, the eye is positioned at the exit pupil. However, in applying projection optics there are tradeoffs to be considered. The exit pupil is technically defined as the image of the aperture stop formed by all lenses following the aperture stop. The aperture stop of the relay system is the relay lens and, therefore, the exit pupil is the image of the relay lens formed by the collimating element, as previously illustrated in figure 3-6.5. Figure 3-6.19 shows the collimating end of the relay system and illustrates the tradeoffs involved in having the eye placed at various positions. At the normal exit pupil position (figure 3-6.19 a), the total field of view may be viewed from any point within the exit pupil diameter $D_x$. The observer has the maximum degree of head freedom with a full field of view instantaneously from all points. However, once the eye is beyond the exit pupil in a lateral direction no field at all exists. It is an all or nothing situation at this boundary. Anywhere within the cross-hatched diamond area, the complete field of view is observable. The length of the base of the diamond is $D_x$ cot $(\beta)$ so this length would cover a wide range when the angle $\beta$ is small. The trade-off involved in having these advantages is that the required diameter for the collimator ($D_c$) is large for a given viewing distance $S_x'$ and field of view. The analyses and trade-off studies investigated in the body of this report are based upon this condition where $D_c = 2 S_x' \sin \beta + D_x$.

The weight and volume of the optics may be reduced for the condition illustrated by figure 3-6.19 b, where the eye is located at the end of the diamond. This is a limiting case of (a) and the trade-off curves with $D_x = 0.5$ are usable for this condition (although $D_x = 0$ in figure 3-6.19 b, the approximation is valid). At that point along the optical axis, the total field of view is available; however, in moving up or down a partial field of view is available. The field is reduced gradually to zero as one moves out to the marginal rays.

However, the diameter over which some of the field may be viewed is large. The diameter of the collimator for this same viewing distance and viewing angle is

$$D_c = 2S_x' \sin \beta + 0.5 \text{ (for a nominal small } D_x \text{ of 0.5)}$$

which results in significant savings in volume and weight.

The third case is illustrated in figure 3-6.19 c, where the eye is located a distance beyond the diamond area so that the full field is not available at any point in this region. Head motion is required to cover the field. In order that the full field of view may be covered, head motion equivalent to nearly the lateral diameter to the marginal rays is required. The diameter of the collimator in this case is the least of the three cases covered, and is determined from

$$D_c = 2S_x' \sin \beta - 2H$$

where $2H$ is the diameter over which eye motion is required to see the field of view. Since the collimator diameter is minimum, so is the volume and weight. This is gained at the expense of head motion and a partial field of view at all times.

The trade-off study for volume and weight is investigated for the case with the eye positioned at the exit pupil of the system in figure 3-6.19 a or b. This results in large volume and weight. The case where $D_x = 0.5$ inches is essentially the condition for zero head motion with the total field of view observable.

To determine the minimum volume and weight which would result for a direct system using eye motion, the following procedure is used.

Figure 3-6.8 is used to establish the collimator diameter for the specified viewing distance $S_{x_0}'$, total field of view, and allowable head motion. Using this collimator diameter, and a head motion of 0.25 inch $\dfrac{(D_x = 0.5)}{2}$ ,

determine an "effective" viewing distance $S_{x_1}'$ for the originally specified field of view. This effective viewing distance is used with an exit pupil of 0.5 inch (figure 3-6.10) at the specified field of view to obtain system volume and weight.

The instantaneous field of view is obtained for these conditions from figure 3-6.8 for $H = 0$, the collimator diameter previously determined, and the specified viewing distance, $S_{x_0}'$.

This procedure is not applicable for a relay system.

# 1. Typical Application Examples of Optical Design and Trade-off Charts

Consider an application which requires a 25 degree field of view. Assume there are two choices for viewing distance, 15 inch for a collimator fixed to the spacecraft structure, and 5 inch for an optical system mounted to the pilot's helmet. In the latter case, head attitude sensing would be required for image stabilization and positioning in the real world as described in Appendix D.

A three-inch exit pupil will be assumed for the first system (System A) to permit freedom of eye position for full field of view. Since the second system (System B) is fixed to the helmet, a smaller exit pupil is adequate, say 1 inch to permit eye pupil motion and a small amount of head motion. Optical designs utilizing f/1 collimators and f/0.75 relay lenses (where applicable) will be assumed.

The following procedure will illustrate the computations necessary and the use of design and trade-off charts in defining refractive and reflective optical system characteristics for both direct and relay systems.

The collimator diameter is determined from figure 3-6.8:

For $2\beta = 25^O$, $S_x' = 5''$ and $15''$ and $D_x = 1''$ and $3''$

Collimator Diameter $D_c = 2.1'' + D_x = 2.1 + 1'' = 3.1''$ for System A

$$= 6.5'' + 3 = 9.5'' \qquad \text{for System B}$$

The focal lengths for f/1 collimators $(F_c = D_c)$ are, therefore:

3.1'' for System A

9.5'' for System B

## a. Direct Optical System

The system solutions for a direct optical system will now be obtained, followed by the corresponding solutions for a relay system.

The CRT diameter, $D_0$, is obtained from figure 3-6.7 as a function of collimator diameter and field of view. CRT diameter may also be obtained from figures 3-6.11 and 3-6.12.

$D_0 = 1.3''$ for System A

$D_0 = 4.1''$ for System B

Referring to table 3-7.7, System A would require a 2 inch CRT and System B a 5 inch CRT to furnish the required usable screen diameter.

For a refractive system or a full reflection system, the above solutions are valid. In the case of a spherical combiner system, the viewing distance should be $2 \times F_c$, the radius of curvature of the mirror, to avoid distortion of the real world. In the case of System A, this distance would be 6.2 inches, not the 5 inch selected, and for System B, 19 inches rather than the 15 inch assumed. In both cases, a lower f/no. collimator would be required, increasing the collimator optical design difficulty; or a smaller field of view or exit pupil would be necessary if the assumed viewing distances must be maintained. Both smaller field of view or smaller exit pupil result in smaller collimator diameter and thus smaller focal length and radius of curvature.

If the exit pupil diameter is reduced to 0.33 inch in case A, and 1.00 inch in case B, the full field of view can be seen within these diameters with collimators of 2.5 inches and 7.5 inches diameter respectively, for viewing locations at the centers of curvature. The corresponding CRT diameters required for these shorter focal length collimators (from figure 3-6.7) are 1.1 inches and 3.3 inches for systems A and B.

If the specified viewing distance were such that even a 25 degree instantaneous field of view were not possible, then head motion would be required to see all of the 25 degree field, although something less than 25 degrees would be viewed at any one time. Figure 3-6.8 is used to determine the instantaneous field, and required head motion for this case. For example, if the required field of view were 40 degrees for a 15 inch viewing distance with a 7-1/2 inch f/1 collimator, ±1-1/2 inches of head motion would be required, and the instantaneous field would be 29 degrees.

The previous examples have considered one eyed fields of view, which is appropriate in the vertical direction. In the lateral direction, the eye spacing increases the lateral field in cases where head motion would be required.

The volumes and weights for Systems A and B, both reflective and refractive, are obtained from figures 3-6.11 and 3-2.12.

| System | Weight (lbs.) | Volume (in$^3$) |
|---|---|---|
| A - reflective | Under 2 | 20 approximately |
| - refractive | Under 2 | 20 approximately |
| B - reflective | 6  (4) | 1000 (500) approximately |
| - refractive | 40 (24) approximately | 1000 (500) approximately |

In the case of a spherical combiner system in which a smaller exit pupil must be used, (figure 3-6.10), as previously described, ($D_x$ = 0.5

inch) and figure 3-6.11 would be used for systems A and B. The resulting weights and volumes are shown in parenthesis in the table above when the differences are significant.

b. Relay Optical System

Solutions to the assumed application will now be described utilizing relay optical systems with an f/0.75 relay lens and an f/1 collimator.

Figures 3-6.15 and 3-6.16 may be used to obtain the required CRT usable screen diameter:

System A    $D_O$ = 0.42 inch

System B    $D_O$ = 1.3 inches

From figure 3-6.9, using scales (1) and (8) for CRT diameter and field of view, the value of Q on scale (5) is found to be 1.0 for System A and 3.1 3.1 for System B. The CRT diameter required for System A is so small, it is actually off scale on the nomograph, and a graphical extrapolation was used to obtain Q for this example. The minimum CRT value shown on the nomograph is typical of current tube technology.

The collimator diameters, based on field of view, distance, and exit pupil size, are as determined for the direct system, i.e., 3.1 inches and 9.5 inches for Systems A and B. The focal lengths for the f/1 collimators are also 3.1 and 9.5 inches.

The relay magnification may now be determined:

$$M_R = \frac{F_c}{Q}$$

$$= \frac{3.1}{1.0} = 3.1 \text{ for System A}$$

$$= \frac{9.5}{3.1} = 3.1 \text{ for System B}$$

Figure 3-6.9 may be used to verify this $M_R$ based on the $f_O$/no. obtained from Q and exit pupil size and the relay f/no. of 0.75. This nomograph may also be used to evaluate the effect of changing f/no. on exit pupil size and on the variation of CRT diameter with field of view for a given exit pupil and $f_O$/no., as described earlier in this section and in Appendix F. The high relay magnification (3.1) determined above results in small CRT size, but long CRT to collimator distance.

Volumes and weights are now obtained from figures 3-6.15 and 3-6.16.

| System | Weight (lbs) | Volume (in$^3$) |
|---|---|---|
| A - reflective | 1 | 40 |
| - refractive | 3 | 60 |
| B - reflective | 16 | 1300 |
| - refractive | 60 | 1900 |

A comparison of the weights and volumes for the direct and relay systems indicates that for the conditions chosen, the direct system is of lighter weight and smaller. The direct systems require larger CRT's, however, although they are stock sizes available in high brightness versions. If the installations did not require a separated collimator and CRT (as in figure 3-6.3 a) and thus a relay lens (projection) system, the direct system would be the application choice from a size and weight viewpoint. However, other considerations, such as additional power for the larger CRT and larger CRT size for the direct system, might affect the trade-off for a specific application.

The previous discussion for a direct spherical combiner systems, which requires an eye position at the center of curvature, is applicable here as well. A reduced exit pupil is necessary for this condition and lower volumes and weights result.

If an f/2 collimator and f/1.5 relay lens were considered for the application example chosen, reference to figure 3-6.18 illustrates the fact that the higher f/no. relay system cannot meet the application requirements.

An f/2 direct system would require 2.7 inch and 8.2 inch CRT screen diameters for systems A and B, respectively, compared with the 1.3 and 4.1 inch CRT's for f/1 system. Viewing distances for a spherical combiner system become 12.4 and 38 inches for Systems A and B, with the eye at the center of curvature, requiring severe restrictions in field of view, or large head motion to observe the specified field of view.

## H. GENERAL APPLICATION CONSIDERATIONS

The following discussion is included to review and clarify the significance of the various optical parameters as they are involved in the application of projection display systems. Both direct and relay systems will be covered.

## 1. Direct System

All the trade-off charts in this section use the term exit pupil as the diameter within which the specified total field of view is visible. The collimator is, in general, the field stop for the CRT image, assuming that the combining mirror is larger than the lens. That is, the collimator limits the amount of field visible for a given eye position. Similar to looking through a window, the field of view gets larger as the observer gets closer to the window or field stop. Also, each eye sees a different field, with both eyes seeing the over-all field. By moving horizontally or vertically, the observer shifts the field of view, thus increasing the over-all field that can be observed. The lateral eye spacing thus increases the lateral field of view over that in the vertical direction. For close-in viewing distances, the collimator ceases to be the field stop and the full image field is seen, even with some head motion.

The diameter of the collimating lens is a function of the field of view, the distance from the pilot's eye, and head motion required without field of view restrictions. The viewing distance should be as short as possible to minimize lens size.

The focal length of the collimating lens is a function of the size of the CRT and the total field of view required. From optical considerations, a large CRT is desired, but for image brightness and packaging, a small CRT has advantages.

In applications requiring dual image sources, such as a CRT and reticle, a larger focal length collimator is desirable to permit space for a beam splitter between the collimator and one of the image sources, to insert the second image.

The lens requirements are somewhat sophisticated for projection display systems, because of the required accuracy and the wide field. In photographic optical systems where the entire aperture is used, vignetting is normally employed at large field angles to remove the poorly corrected rays. This cannot be done in projection display systems without reducing the instantaneous field of view for off-axis eye positions. This would then require additional head motion to allow observance of the entire field of view. In general, then, the lens requirements are in some respects more severe than for a photographic system and in other respects (as in resolution) less severe.

## 2. Relay System

The primary advantage of a relay system is the possibility of attaining a large field of view in the type of arrangement shown in figure 3-6.3a. The collimator is converted to a combiner by using a spherical surface. The

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concave side of the glass is made partially reflecting so that it acts as a concave mirror to collimate the CRT image. The spherical combining glass is designed with the pilot's eye near the center of curvature and with both surfaces having the same center. This arrangement reduces to a minimum the distortion of the outside world.

In this system, a relay lens projects an aerial image of the CRT face on the focal surface of the collimating mirror. The collimating mirror images the aerial image at infinity and super-imposes it on the outside world. At the same time, the collimating mirror images the aperture of the relay lens forming an exit pupil at the pilot's eye location. With this arrangement, the pilot can see the total field of view with both eyes without the necessity of head motion, as long as his eyes are in the exit pupil.

Regarding the effect and utility of this exit pupil: it is a real image of the exit aperture of the projection lens, acting as a large "knothole" in an invisible "fence". Through it the entire field of projected images is visible, provided only that the eye is near enough so that it does not act as a stop. Yet the exit pupil offers no impediment to the pilot's vision or head motion. Thus, it can be placed at, or immediately in front of his face. Fore and aft head motion reduces the effective dimensions of the "knothole" for extreme field rays. In the plane of the "fence", or exit pupil, lateral or vertical eye motion will produce a sharp cutoff of the viewed images at the edges of the pupil. Fore and aft of the plane, the cutoff is "soft" - some extreme field angles disappear first, while the opposite extreme angles remain in view longest (figure 3-6.4).

There are several advantages in the relay system approach for a wide-angle viewing system. The diameter of the optical projector required depends on the diameter of the exit pupil desired and on the location of the exit pupil relative to the collimator focal plane. Very wide fields of view can be obtained with relatively small diameter optics (excluding the reflecting combining lens, which need be no larger than a corresponding flat combiner). Glass parts can be kept at a reasonable distance from the pilot's face, and the projector can more easily be removed from his normal visual field.

The magnification between the CRT face ($D_o$) and the aerial image ($D_i$) is another feature that gives more flexibility in the design of relay systems. In effect, this allows use of a smaller, brighter CRT.

There are, however, difficulties in the design of a spherical combiner relay system. Most of these result from the optical elements not having a common optical axis. (An additional beam splitter in the pilot's vision line

would make it on axis; however, this solution is impractical for most installations.) The off-axis arrangement in figure 3-6.3 a results in image errors. A maximum included angle $\theta$ (figure 3-6.3) of 30 degrees is a practical limit for reasonable lens designs to minimize such errors.
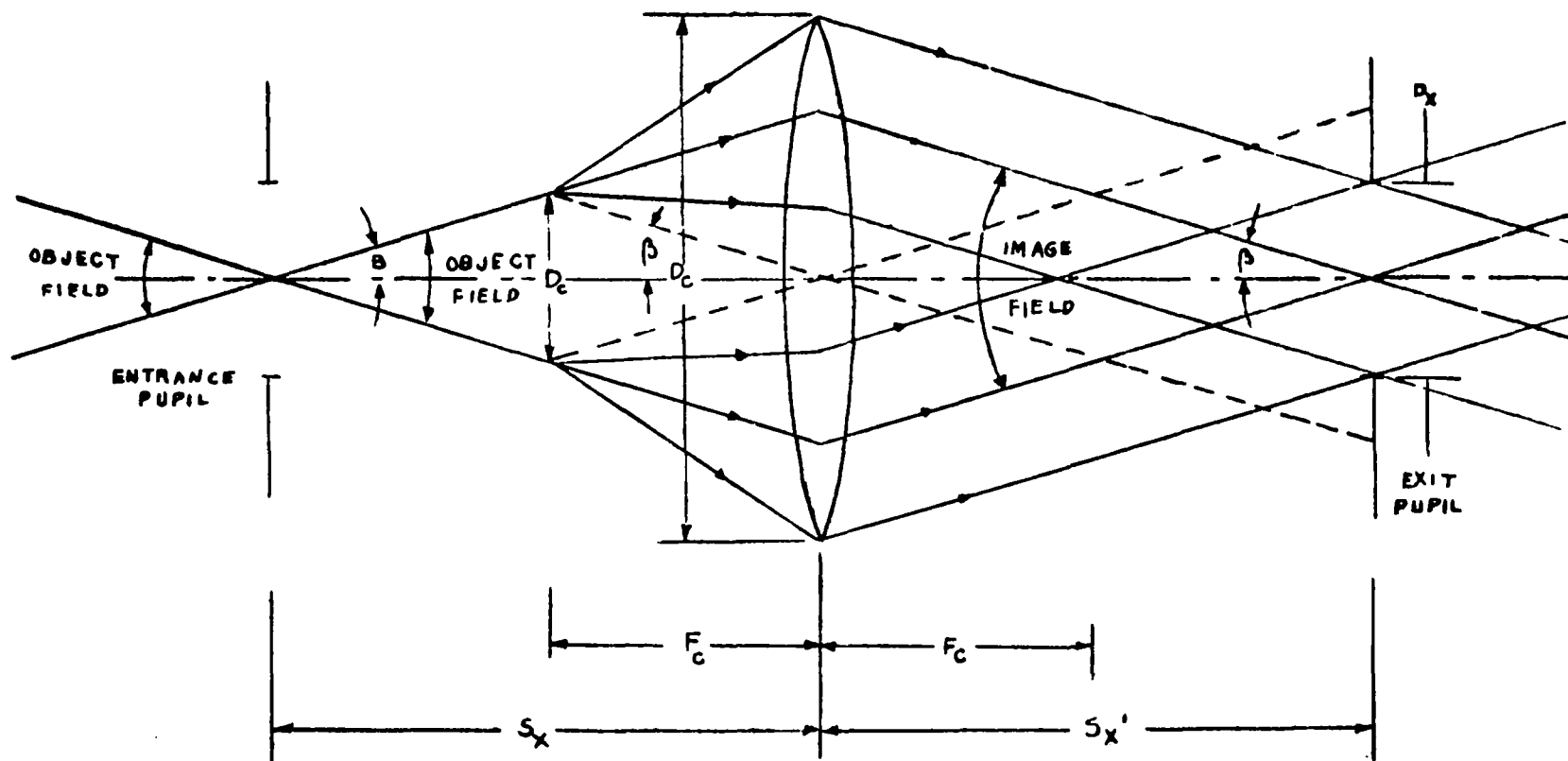
FIGURE 3-6.2
DIRECT REFRACTIVE SYSTEM

Relay lens

CRT

Partially reflective combiner

θ

eye

(a)

Fully reflective collimating mirror
beam splitter

eye

relay lens

CRT

(b)

beam splitter

eye

collimating lens

relay lens

CRT

volume
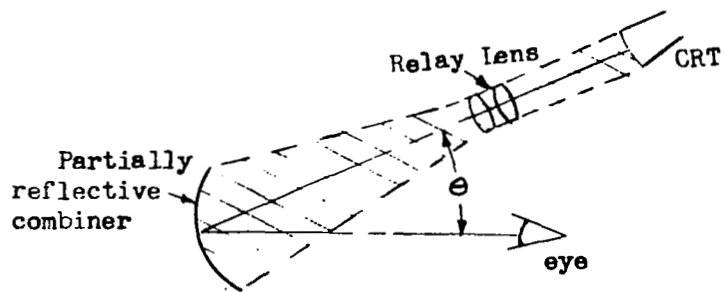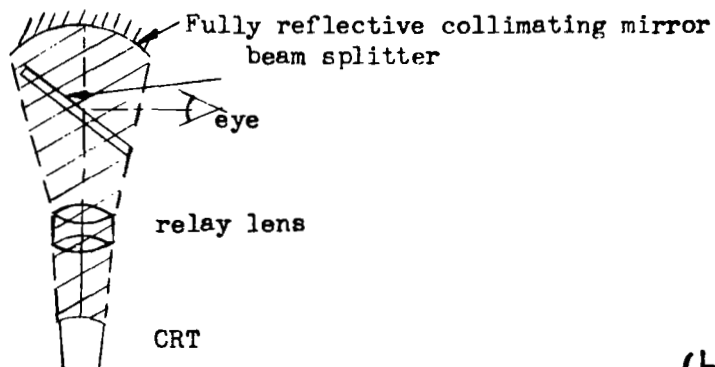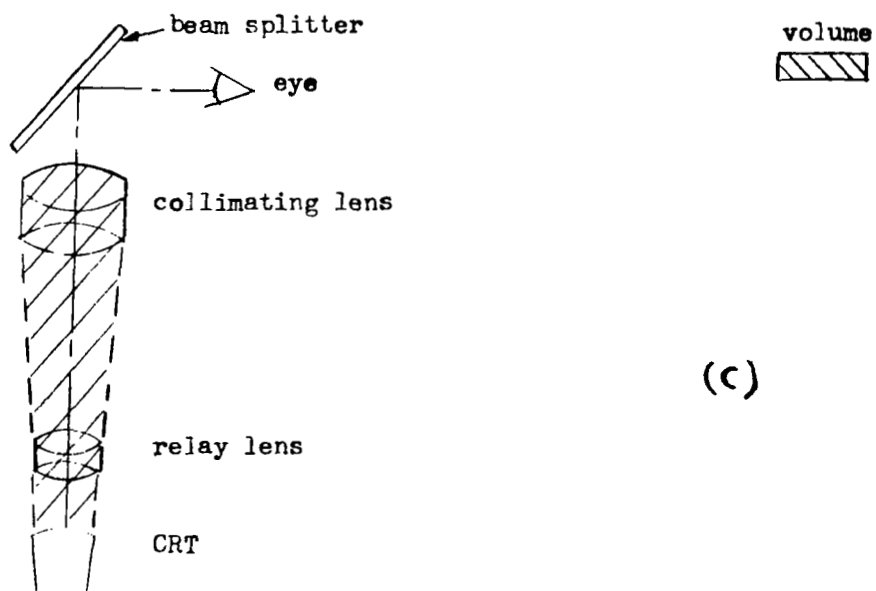
(c)

FIGURE 3—6.3
RELAY OPTICAL SYSTEM CONFIGURATIONS

USEABLE DIAMETER

CRT

PROJECTION
ELEMENT
RELAY LENS
(ENTRANCE PUPIL)

IMAGE PLANE

COLLIMATING ELEMENT
(REFLECTIVE OR
REFRACTIVE)
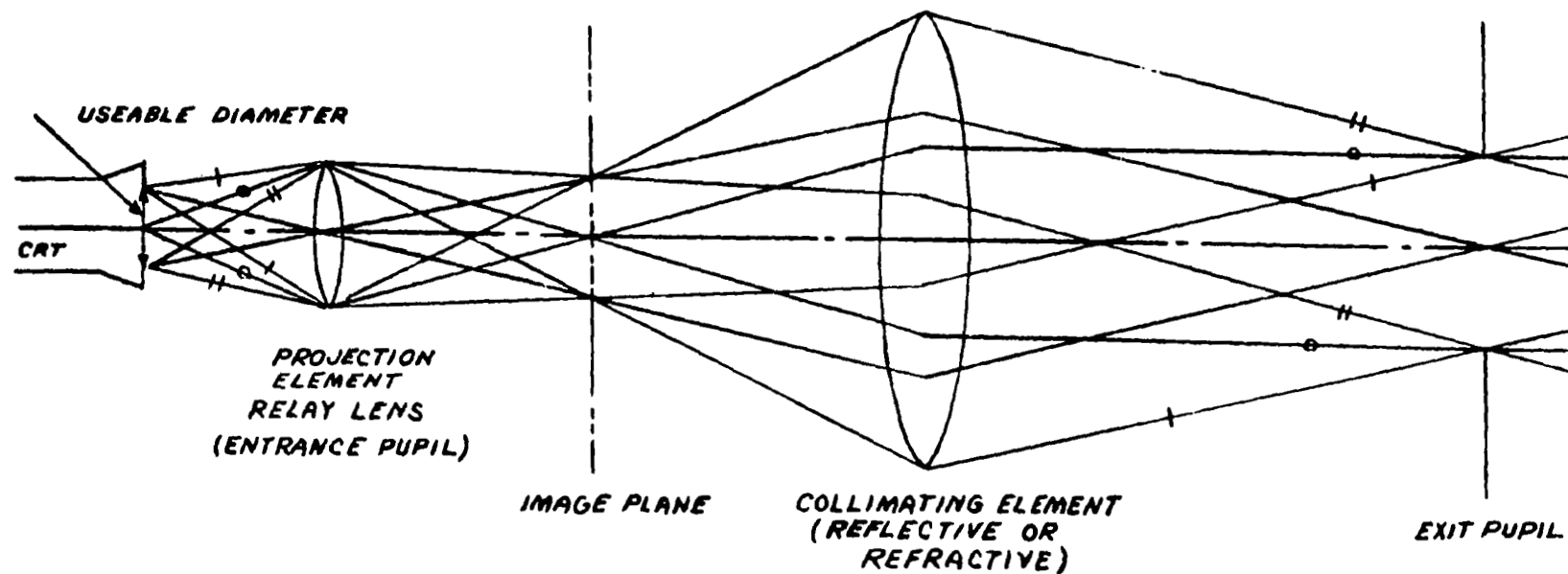
EXIT PUPIL

FIGURE 3-6.4
INTERNAL IMAGE RELAY SYSTEM
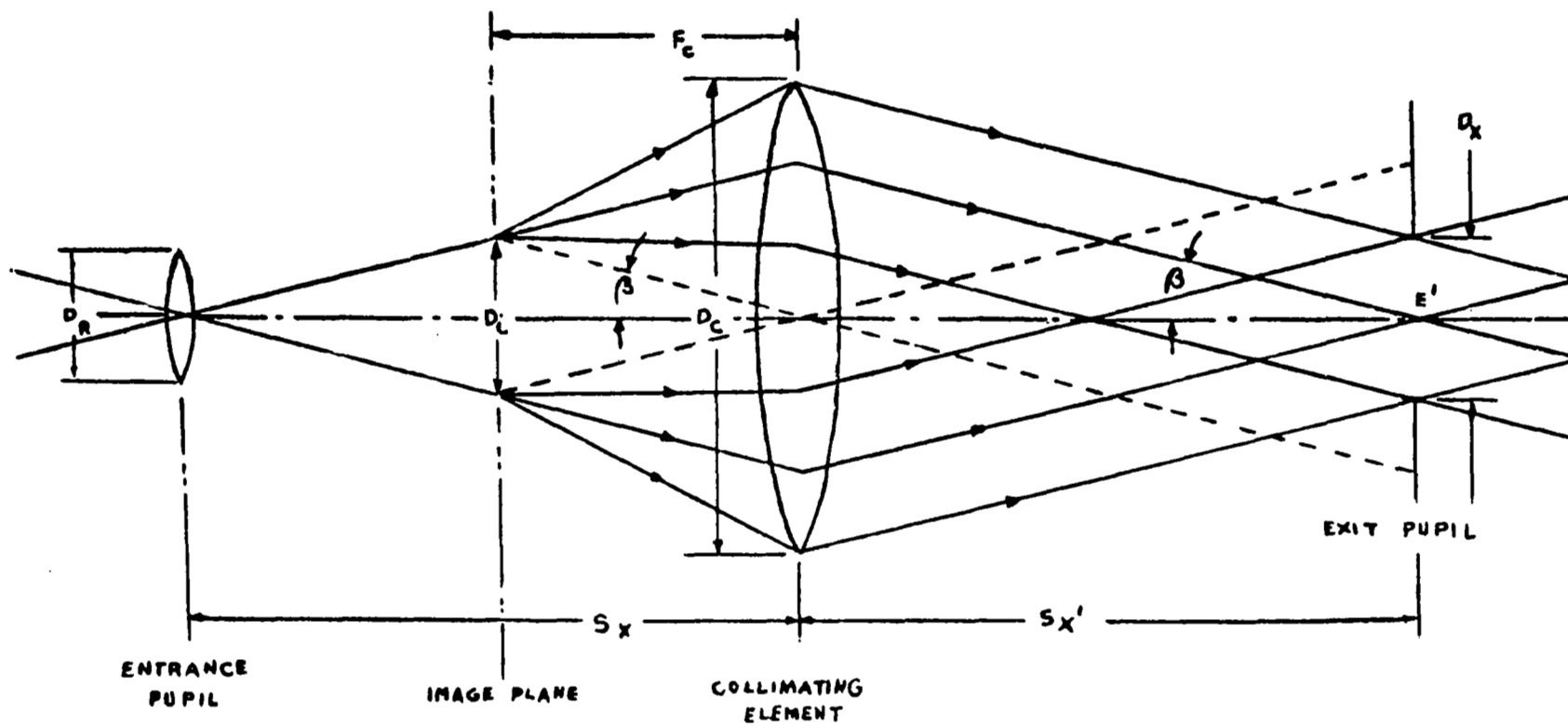
FIGURE 3-6.5

EXIT PUPIL GEOMETRY FOR INTERNAL IMAGE RELAY SYSTEM
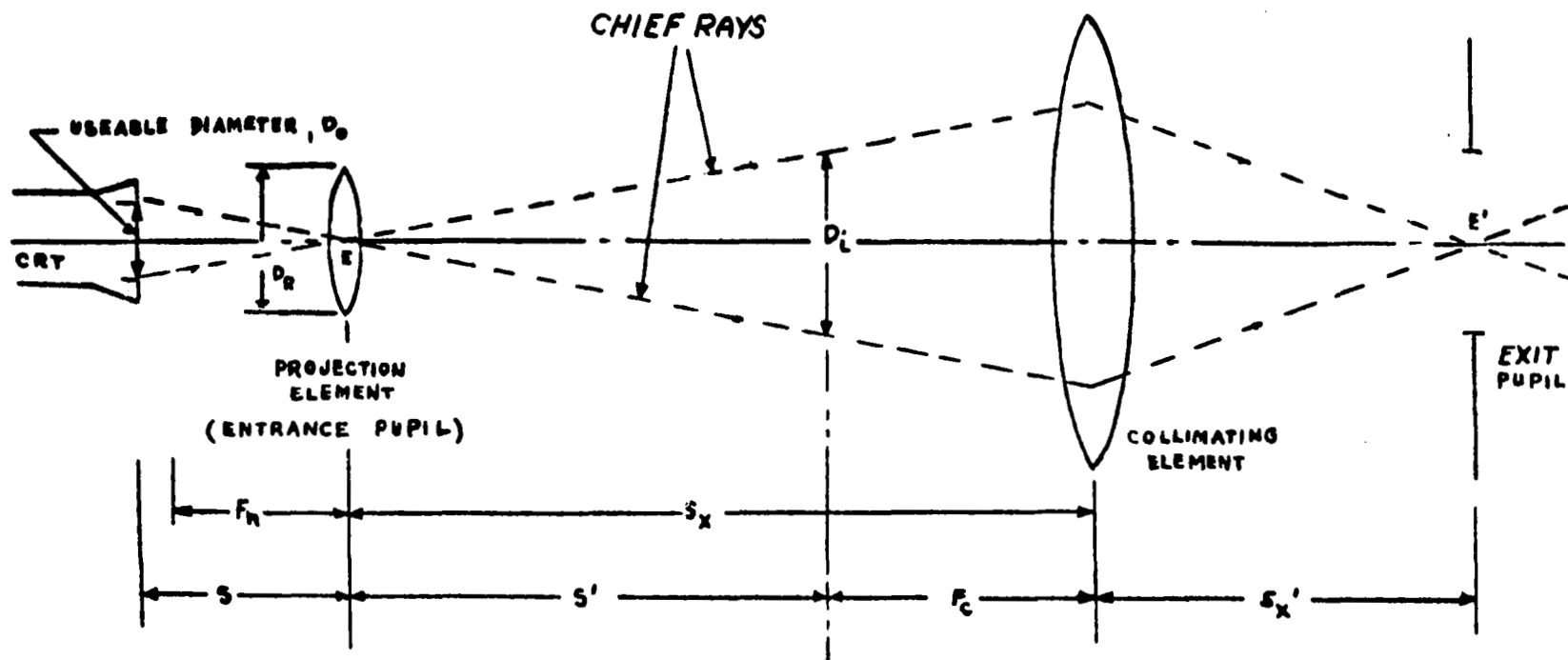
FIGURE 3-6 6
INTERNAL IMAGE RELAY SYSTEM GEOMETRY

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## SPHERICAL MIRROR SYSTEM

SPHERICAL MIRROR
PLANE COMBINER
EYE
CRT

PLANE COMBINER
SPHERICAL COMBINER
EYE
R
CRT

## SPHERICAL COMBINER SYSTEM (EYE AT CENTER OF CURVATURE)
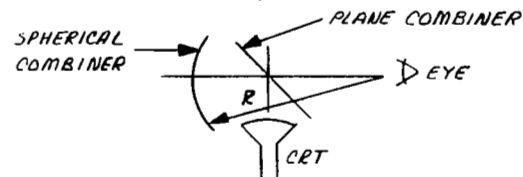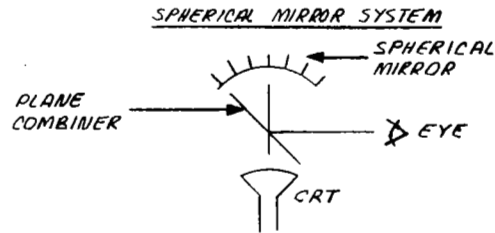
## EYE MOVEMENT CURVES

EYE AT FOCAL PLANE (SPHERICAL MIRROR)

EYE AT CENTER OF CURVATURE (SPHERICAL COMBINER)

|  | EYE AT FOCAL PLANE | | EYE AT CENTER | |
|---|---|---|---|---|
|  | f/1 | f/2 | f/1 | f/2 |
| COLLIM DIA. | D | D | D | D |
| VIEW DIST. | D | 2D | 2D | 4D |
| FOC. LENGTH | D | 2D | D | 2D |

NOTES:

1. USE THIS FIGURE FOR CRT SIZE, COLLIMATOR DIA, FIELD OF VIEW FOR REFLECTIVE AND REFRACTIVE SYSTEMS.

2. SEE FIGURE 3-6.8 FOR EFFECT OF COLLIMATOR SIZE ON FIELD OF VIEW FOR REFRACTIVE SYSTEMS AT ALL VIEWING DISTANCES.

3. USE f/1 CHART WITH COLLIMATOR FOCAL LENGTH = COLLIMATOR DIAMETER TO OBTAIN IMAGE FIELD FOR ANY CRT DIA. AND COLLIMATOR FOCAL LENGTH.

### f/1 COLLIMATOR

COLLIMATOR DIA, $D_c$ — INCHES

TOTAL FIELD OF VIEW — DEGREES $2\beta$

CATHODE RAY TUBE DIAMETER, $D_0$ INCHES

### f/2 COLLIMATOR

$D_c = $ INCHES EYE MOVEMENT
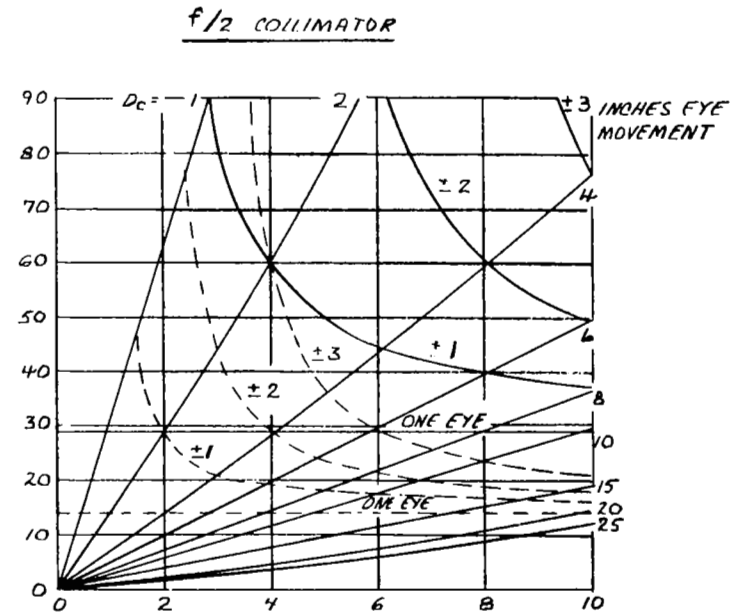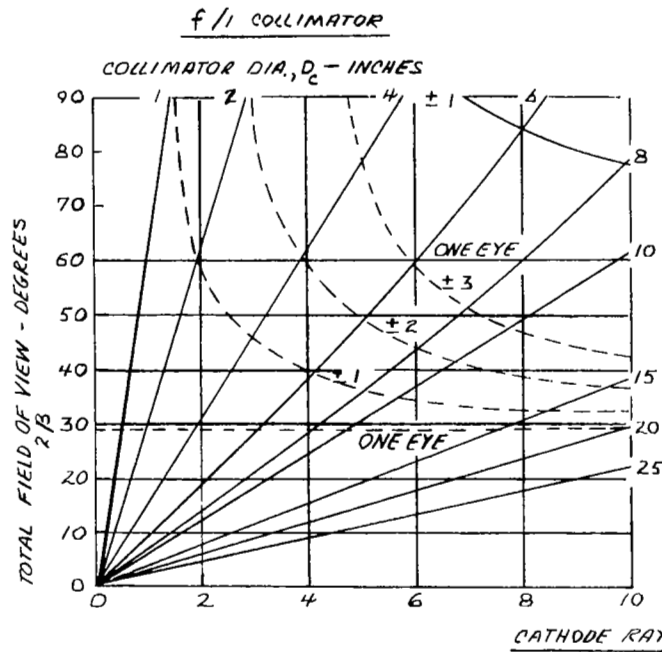
**FIGURE 3-6.7**

FIELD OF VIEW AND CATHODE RAY TUBE DIAMETER FOR DIRECT OPTICAL SYSTEMS
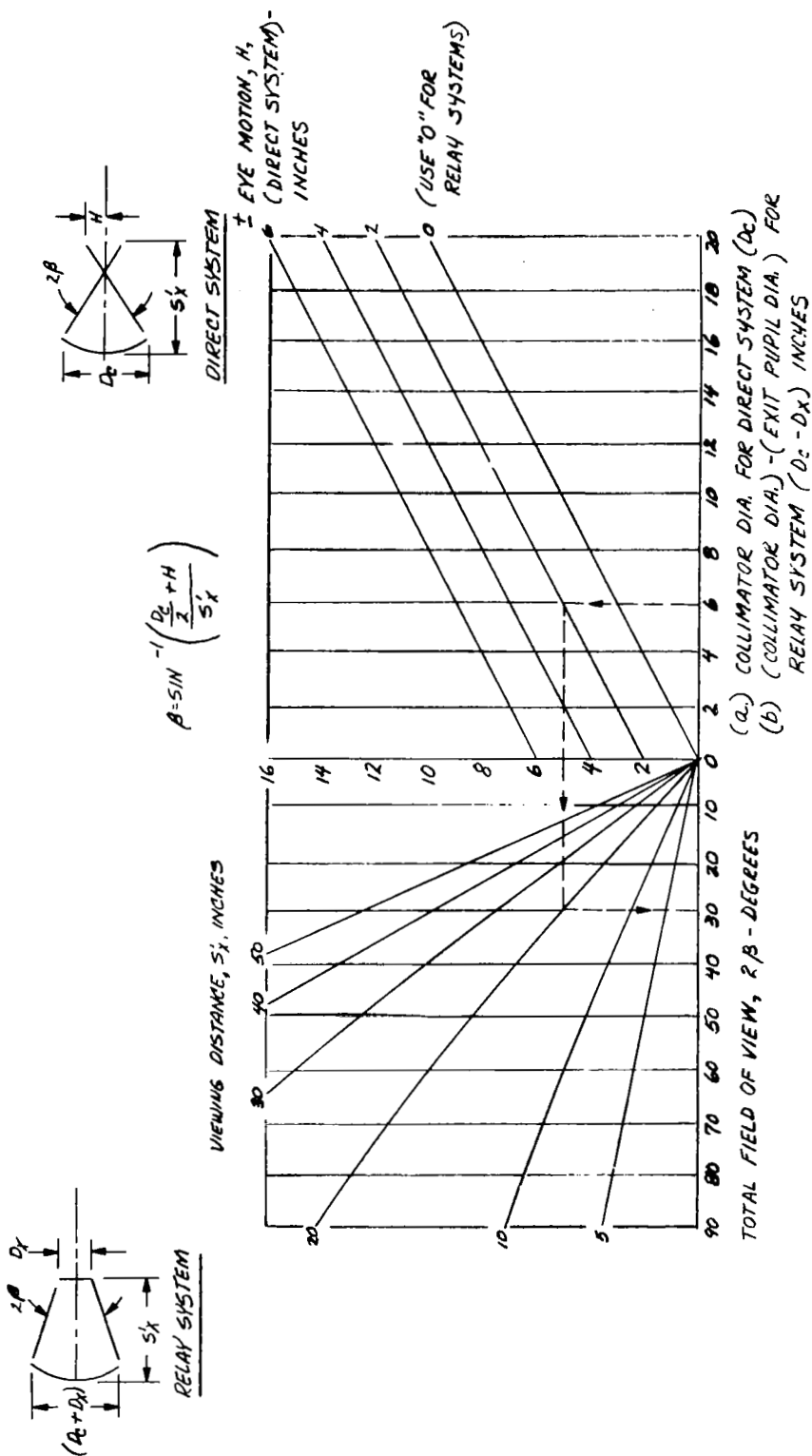
FIGURE 3—6.8

FIELD OF VIEW, COLLIMATOR DIAMETER, VIEWING DISTANCE AND EYE MOTION FOR REFRACTIVE
COLLIMATORS AND FULLY REFLECTIVE MIRROR COLLIMATORS FOR BOTH DIRECT AND RELAY
SYSTEMS
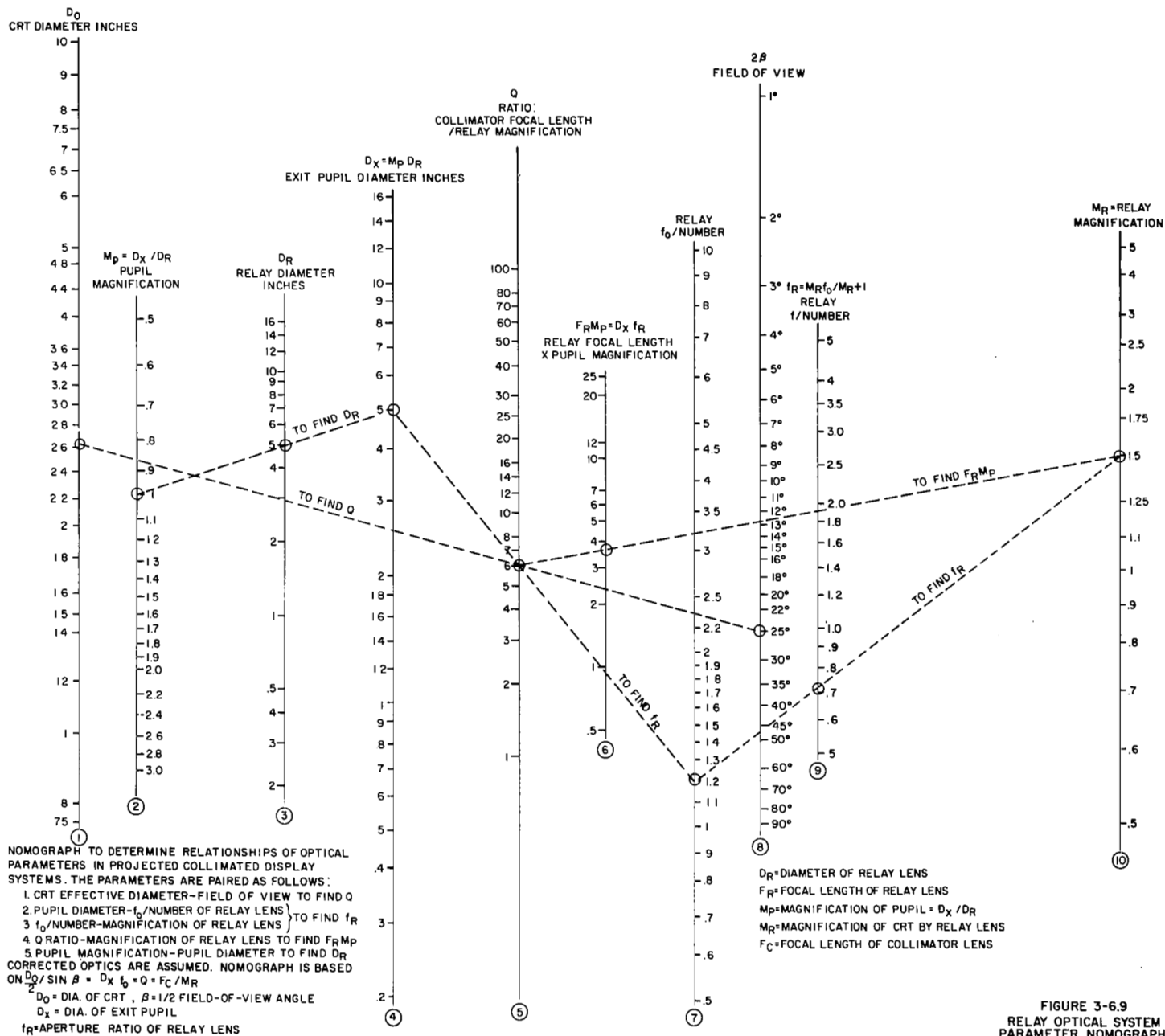
FIGURE 3-6.9
RELAY OPTICAL SYSTEM
PARAMETER NOMOGRAPH
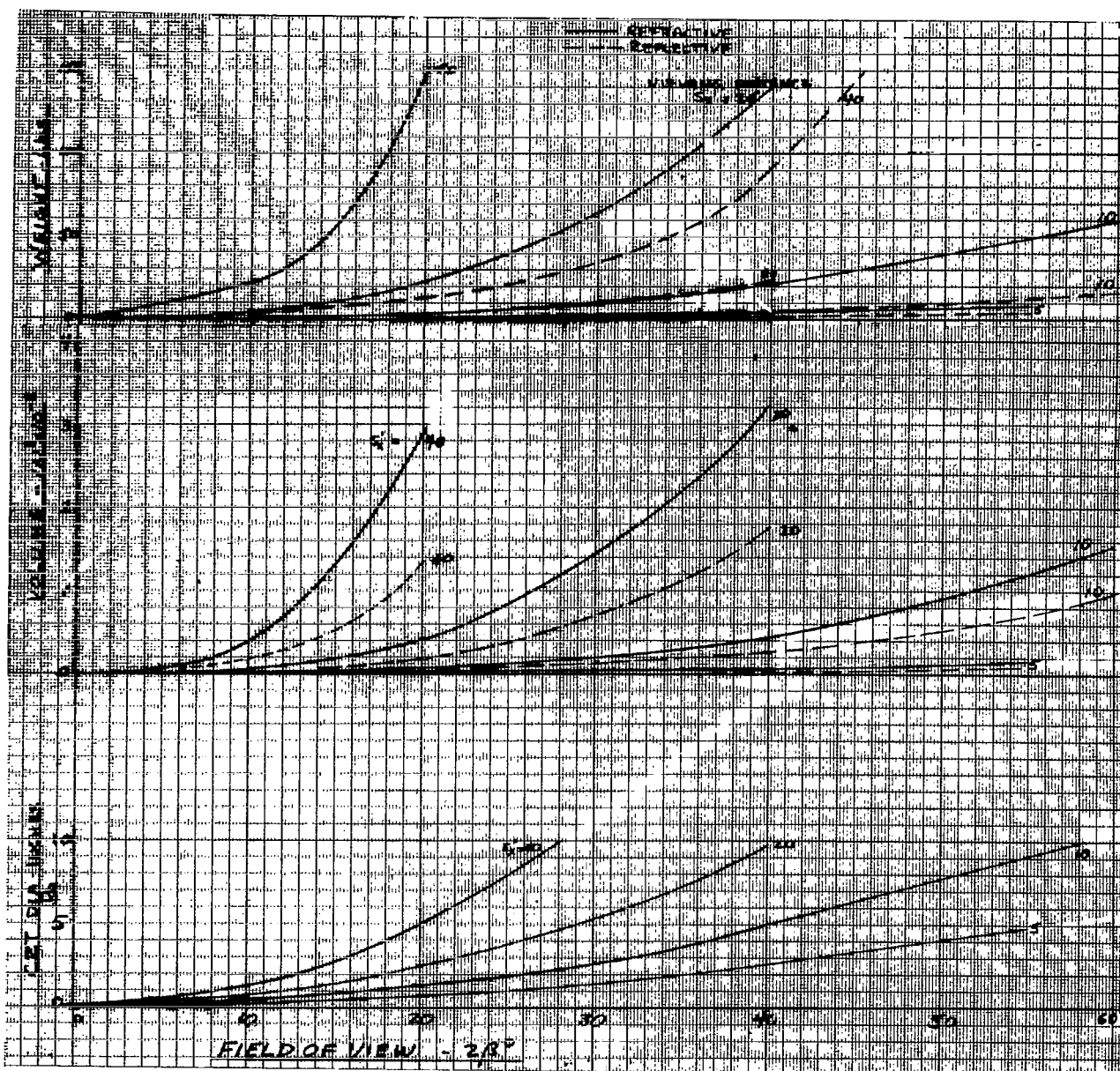
FIGURE 3-6.10
DIRECT OPTICAL SYSTEM
WEIGHT-VOLUME TRADEOFF
f/I COLLIMATOR — EXIT PUPIL DIA=0.5 INCH

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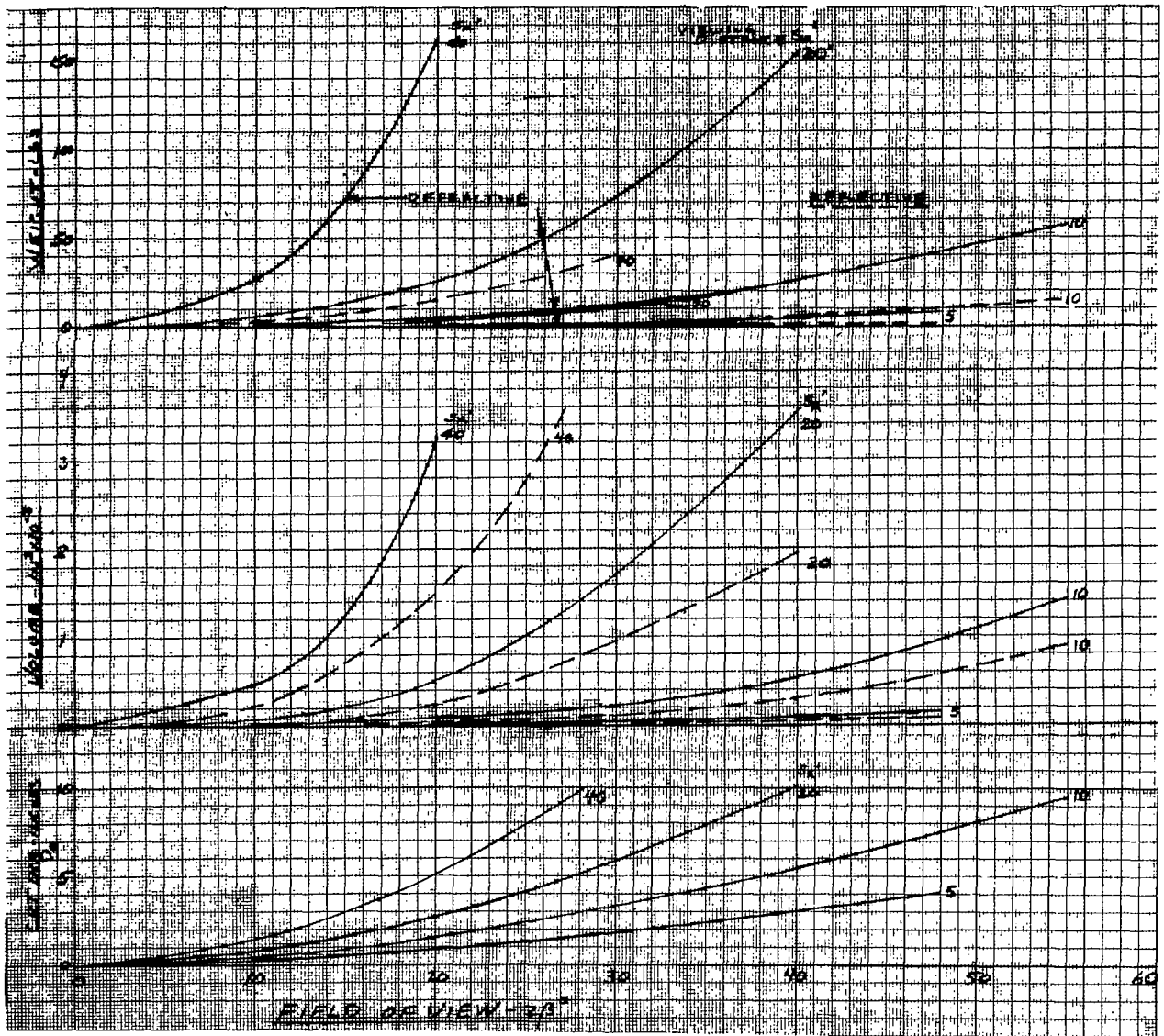
FIGURE 3-6.11 DIRECT OPTICAL SYSTEM
WEIGHT-VOLUME TRADEOFF
f/I COLLIMATOR — EXIT PUPIL DIA. = I INCH

130
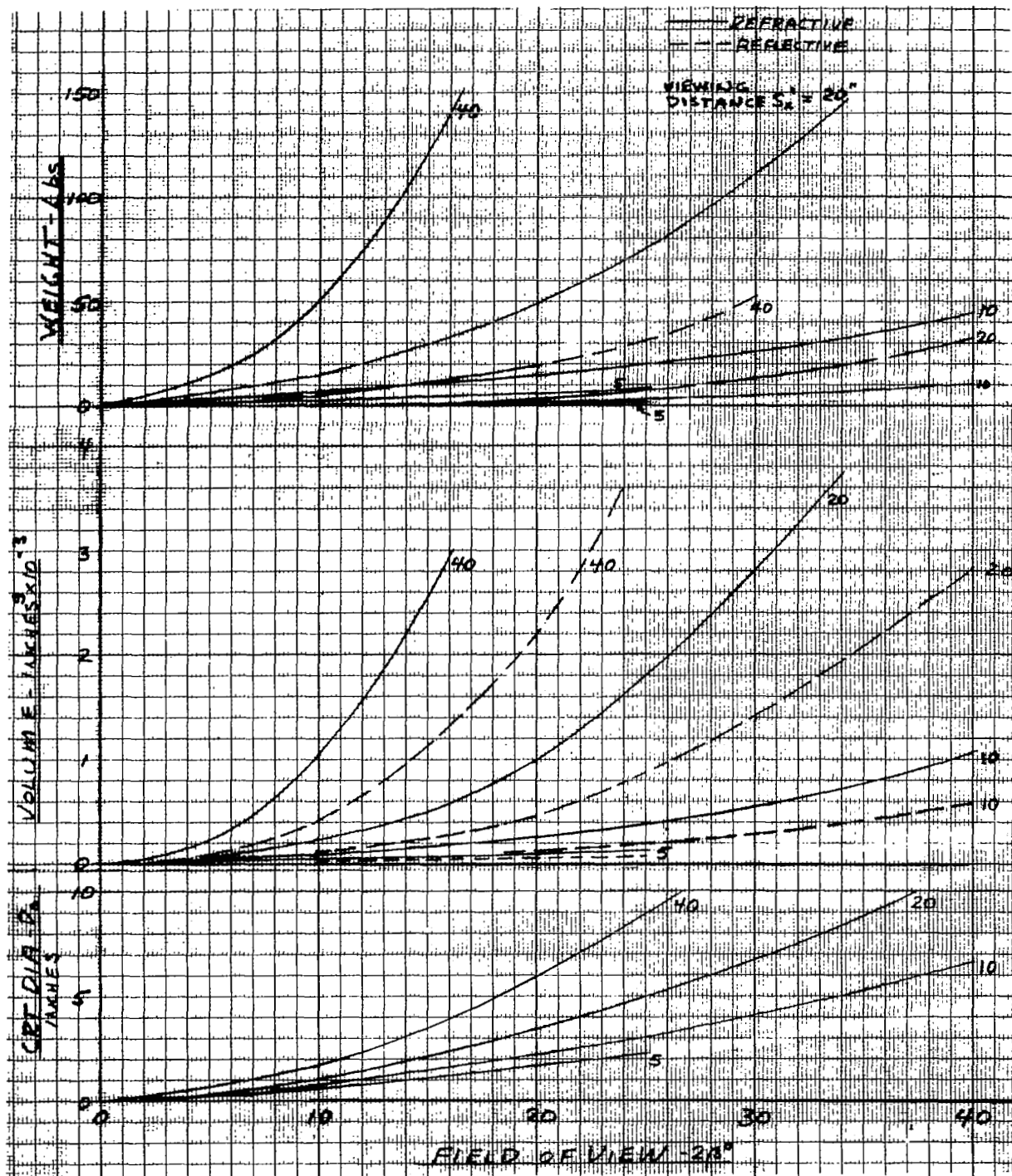
FIGURE 3-6.12. DIRECT OPTICAL SYSTEM
WEIGHT-VOLUME TRADEOFF,
f/1 COLLIMATOR — EXIT PUPIL DIA. ≅ 3 INCHES

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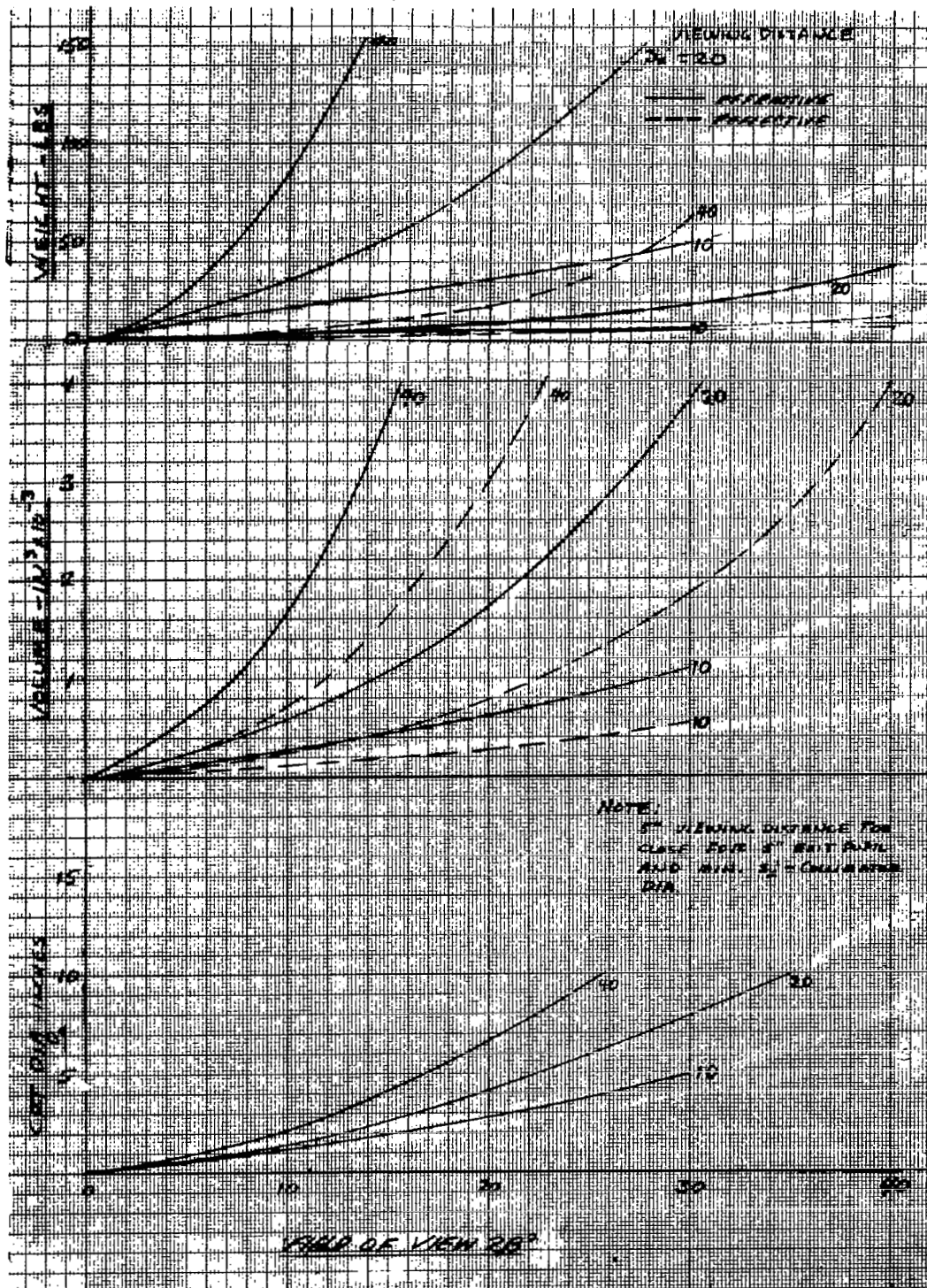
FIGURE 3—6.13
DIRECT OPTICAL SYSTEM
WEIGHT—VOLUME TRADEOFF,
f/1 COLLIMATOR —
EXIT PUPIL DIA =5 INCHES
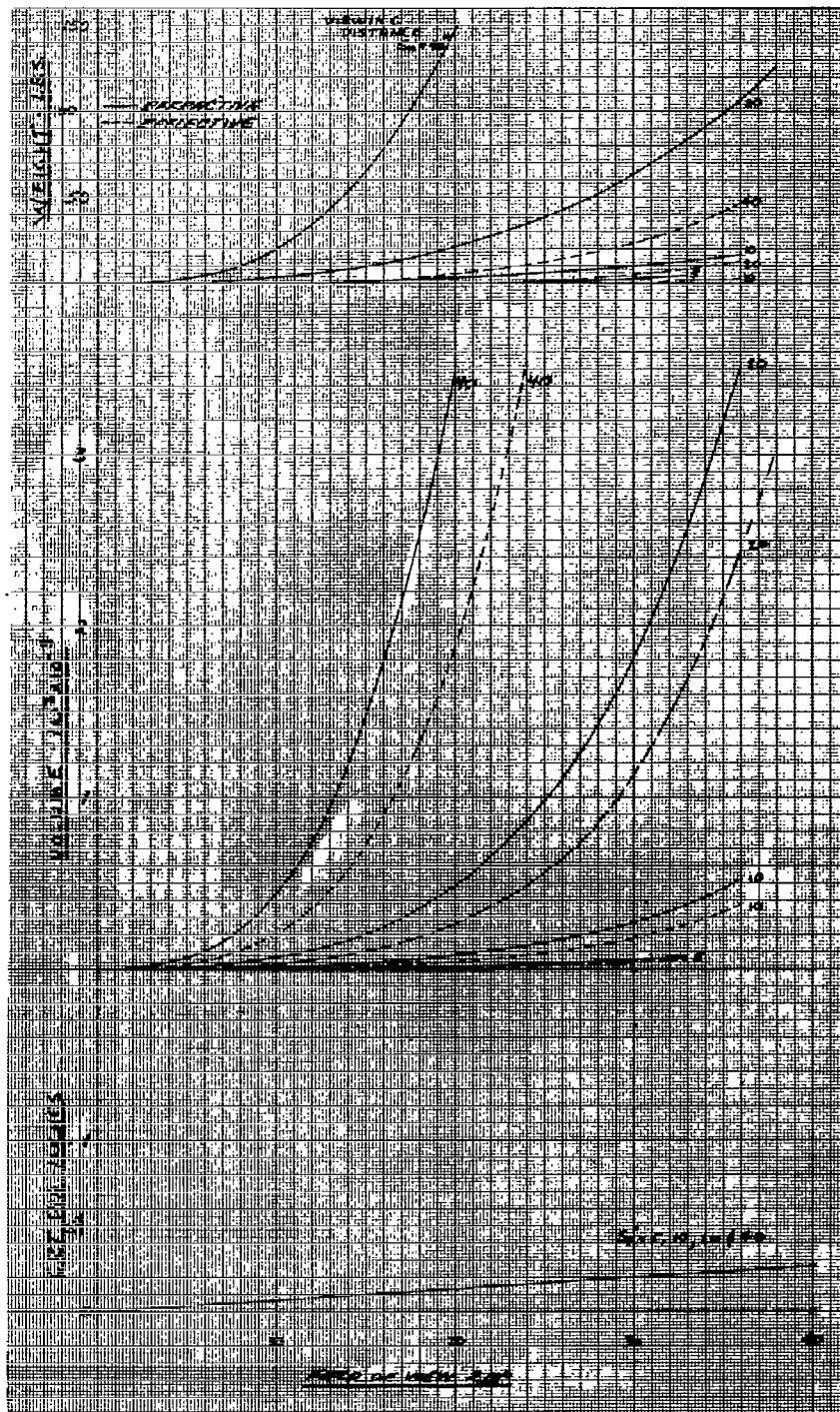
FIGURE 3-6.14
INTERNAL IMAGE RELAY SYSTEM
WEIGHT-VOLUME TRADEOFF,
f/1 COLLIMATOR AND f/O .75 RELAY
LENS — EXIT PUPIL DIA.=0.5 INCH

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FIGURE 3-6.15
INTERNAL IMAGE RELAY SYSTEM
WEIGHT - VOLUME TRADEOFF, f/1
COLLIMATOR AND f/0.75 RELAY
LENS - EXIT PUPIL DIA. = 1 INCH
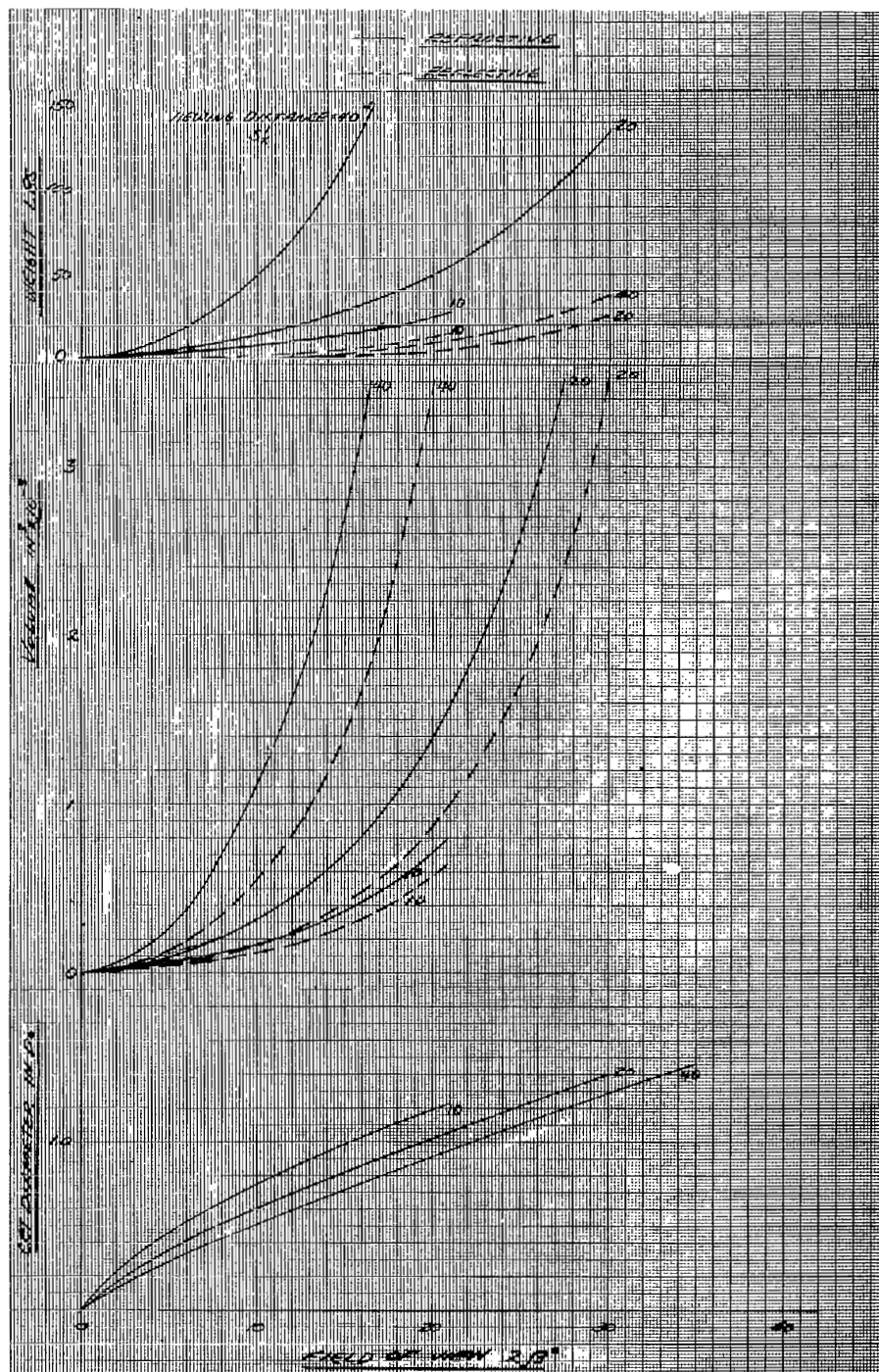
FIGURE 3-6 16
INTERNAL IMAGE RELAY SYSTEM
WEIGHT-VOLUME TRADEOFF, f/1 COLLIMATOR
AND f/0.75 RELAY LENS — EXIT PUPIL DIA.=3 INCHES
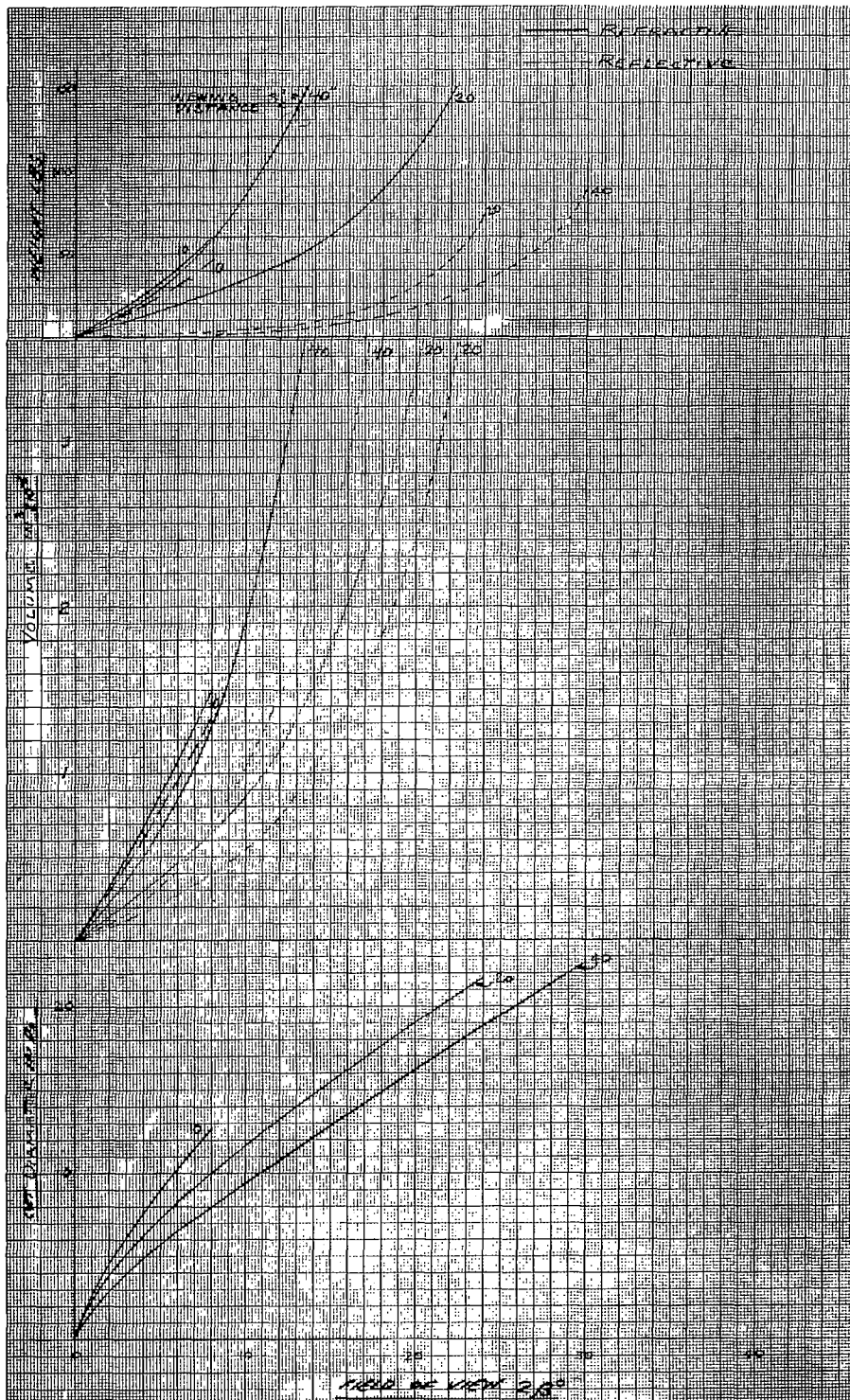
FIGURE 3-6.17
INTERNAL IMAGE RELAY SYSTEM
WEIGHT-VOLUME TRADEOFF f/1 COLLIMATOR
AND f/0.75 RELAY LENS—EXIT PUPIL DIA.=5 INCHES

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FIGURE 3-6.18
INTERNAL IMAGE RELAY SYSTEM
WEIGHT-VOLUME TRADEOFF f/2 COLLIMATOR
AND f/1.5 RELAY LENS — EXIT PUPIL DIA.=3 INCHES
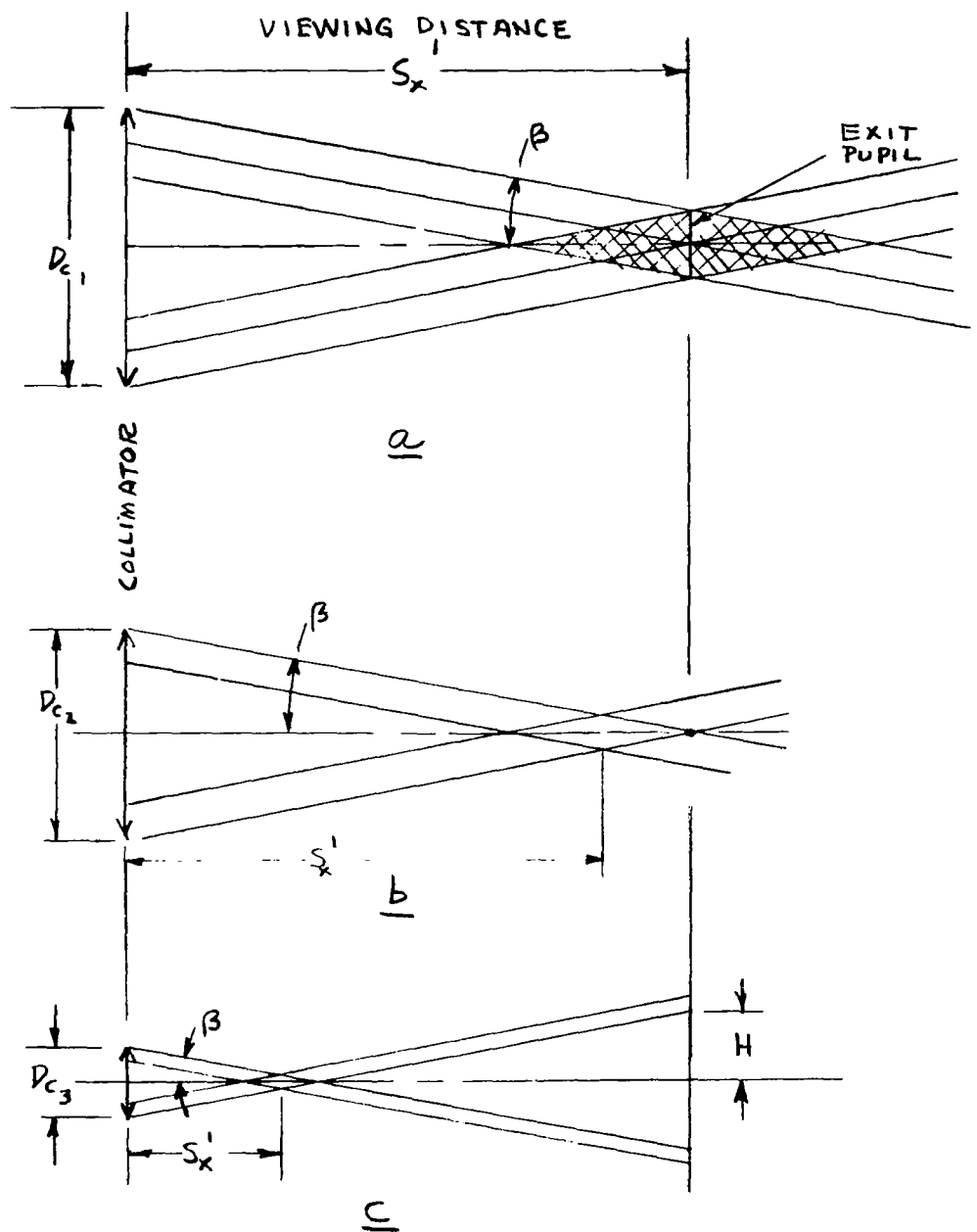
FIGURE 3-6.19
COMPARISON OF EYE MOVEMENT, EXIT PUPIL AND COLLIMATOR DIAMETER RELATIONSHIPS

## 3-7. ELECTRONICS TRADE-OFF STUDY

### A. DESCRIPTION OF SYSTEMS CONSIDERED

The systems considered in this report, as described earlier, are the following:

1. System No. 1 - Simple flight path control display which includes an additional symbolic element. The path control element is positioned against the visually perceived background, such as the rendezvous target, for control.

2. System No. 2 - A dual channel and dual color display system which generates a control and situation display. Situation elements are shown as simple cues such as dots, circles, etc.

3. System No. 3 - Same as System No. 2, except that situation elements will show true size, shape and perspective.

System No. 1 comprises two servos for positioning a mechanical reticle vertically and laterally on the basis of elevation and lateral command information. In addition, a 500 microampere meter movement is appended to the reticle and may be driven directly by a second d-c command signal. All inputs are assumed dc, as received from a sensor or central computer facility, and the servos are dc. The latter will employ brushless d-c motors, several types of which are available and offer improved reliability. The d-c servos were selected for compatibility with spacecraft anticipated d-c signal inputs.

System No. 2 generates and projects, via a CRT and reticle, all images and cues required, exclusive of a true perspective situation cue. The latter is added to System No. 2, to produce System No. 3.

The basic character of System No. 2 is similar to other heads-up display systems developed by Sperry under various programs (e.g., reference 4).

These programs, to-date, have utilized conventional military-type components and employ three follow-up servos for pitch, yaw and roll synchro inputs.

Maintaining these features, a one-for-one substitution of microelectronic elements was effected to arrive at a microelectronics version of the display system. Table 3-7.1 shows size, weight and power requirements for the microelectronic versions of Systems No. 1 through No. 3.

The types of containers required are also indicated, with estimated weights and the number of servos and cards contained. The cards are roughtly 1 x 4.5 x 3.25 inches and approximately 7 ounces in weight. No separate controller weights and volumes are shown since these function are assumed integrated in basic spacecraft systems.

The true perspective cue generator is shown in a 1/2 ATR long container which is assumed to contain all the hardware required to generate and project this cue.

Further condensing of circuits and hardware can be effected by employing sterner and extreme techniques for packaging. Total weight can be reduced by eliminating virtually all internal connectors; each mating pair accounts for approx. 1 ounce and for approximately 48 cards, this corresponds to 3 pounds total. Non-metallic structural materials can be employed such as plastics and ceramics and with very careful mechanical designs the net weight can be reduced further. Redesign of portions of the electronics may add to compactness especially when final requirements are defined. Each of these features carries along with it one or more disadvantages. These may range from restricting the versatility and limiting the maintainability to increasing the number and degree of hot spots. Notwithstanding, it is estimated that a net savings of approx. 6 pounds can be made in the Electronics Box and 2 pounds in the Landing Area Generator for a total of 8 pounds. For each of these boxes this constitutes 20 to 25 percent savings; however, considering that System No. 3 weighs approximately 50 pounds (with shock mounts), the overall savings is only approximately 15 percent.

The systems described in table 3-7.1 are altered in various manners to make them compatible with other forms of input signals. In particular, the systems have been analyzed to determine changes required with input data for pitch, yaw and roll in the following forms:

1. Three-wire synchro

2. D-C analog

3. Digital, Binary, parallel.

The changes have been limited to effecting direct substitutions for the servos. Table 3-7.2 gives data on the three servos presently employed and the associated 2-1/2 cards. In addition, it shows what occurs to size, weight, and power requirements when the servos are eliminated. The three forms of input data given above are assumed and the effects are indicated. It may be noted that size, weight and reliability all suffer when departing from the three servos. Also, indicated in the table are several special devices required - these produce binary number outputs in parallel form for either d-c analog input or three-wire synchro data. Approximate prices of the special devices are also shown.

# TABLE 3-7.1

## SUMMARY OF SYSTEMS NO. 1, 2 AND 3
### ELECTRONICS COMPONENT WEIGHTS, VOLUMES AND POWER REQUIREMENTS

| System | No. of Cards | No. of Servos | Estimated Container Weight (pounds) | Total Weight (including container) | Total Volume (excluding container) | Power |
|---|---|---|---|---|---|---|
| System No. 1 | 2 | 2 | 4 | 8.373 lbs. | 106.0 in$^3$ | 30 VA, 0.8 PF lag 115 VAC, 400~, 1∅ |
| System No. 2 (1 ATR LONG) | | | | (42.3 lbs.) | (873.8 in$^3$) | (330 VA) 330 VA, 0.7 PF lag |
| Electronics Box | 46 | 3 | 12 | 40.3 lbs. | 833.8 in$^3$ | 115 VAC, 400~, 1∅ |
| Sighthead | - | - | 3 | (15.4 lbs.) 13.0 lbs. | (243.0 in$^3$) 177.0 in$^3$ | |
| System No. 3 (1 ATR LONG) | | | | (42.3 lbs.) | (873.8 in$^3$) | |
| Electronics Box | 46 | 3 | 12 | 40.3 lbs. | 833.8 in$^3$ | (360 VA) 330 VA, 0.7 PF lag |
| Sighthead | - | - | 3 | (15.4 lbs.) 13.0 lbs. | (243.0 in$^3$) 177.0 in$^3$ | 115 VAC, 400~, 1∅ |
| (1/2 ATR LONG) Landing Area Generator | 24 | - | 6 | 17.0 lbs. | 354.5 in$^3$ | |

NOTES:

  a. When both figures with parenthese and without are shown those in parentheses refer to 2-color (CRT + Reticle) system and the others to 1-color CRT system

  b. System No. 3 = System No. 2 + perspective cue (landing area generator)

  c. Above figures apply to both conventional and microelectronics versions.

TABLE 3-7.2

SUMMARY OF SUBSTITUTIONS FOR PITCH, YAW AND ROLL SERVOS
AS A FUNCTION OF FORM OF INPUT DATA

| System | Assy's No. of Servos | No. of Cards | Total Volume Excluding Container (cu. in.) | Total Weight Excluding Container (lbs.) | Additional Container Req'd | Total Weight Including Added Container & Tray (lbs.) | Power Req'd (Watts) | Special Devices Req'd | Weight of Special Devices (lbs.) | Volume of Special Devices (cu. in.) | Cost of Special Devices | Failure Rate %/1000 Hrs |
|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 3-Wire-Synchro 3-Servo System | 3 | 2-1/2 | 168 | 5.2 | - | - | 30 | - | - | - | - | 9.2 |
| 3-Wire-Synchro No-Servo System | - | 13 | 795.8 | 20.5 | - | - | 37.6 | 1 Towson Labs. Encoder (ID116) | 7.0 | 300 | $18,000 | |
| | | | | | | | | 1 Towson Labs Capcoder (ID115) | 7.0 | 300 | $18,000 | 27.7+ |
| D-C Inputs (Angle Analogs) | - | 19-1/2 | 734.7 | 19.0 | 1/2 ATR Rack + Harness, Tray | 25.0 | 70.0 | 3 Towson Labs Mod. OC1501 Converters | 10.5 | 450 | $27,000 | 52.6+ |
| Binary Parallel Inputs (All-Angle Data) | - | 23 | 785.8 | 20.6 | 1/2 ATR Rack + Harness, Tray | 26.5 | 84.0 | 3 Towson Labs. Mod. OC1501 Converters | 10.5 | 450 | $27,000 | 63.0+ |

NOTES:

1. Towson Labs. units do not have moving parts
2. Above figures apply to conventional & microelectronic versions
3. Towson Labs. units come in their own containers, with connector assy's
4. Above figures are only for the servos & associated servo amps. & power supplies (or their substitutes).

*Reduce and insert sideways on page 142.*

Further modifications have been contemplated to the extent of altering size of cards, employing magnesium construction rather than aluminum and abbreviating the functions performed. These considerations give rise to four tables, 3-7.3, 3-7.4, 3-7.5 and 3-7.6.

Table 3-7.3 Systems No. 1 through No. 3 employing large cards, 4.5 x 5.5 x .8 inches rather than 3.25 x 4.5 x 1 inches; magnesium construction where possible; magnesium containers; servos used and full functions performed. Full functions denotes that all required computations are performed within the systems and none are relegated to an external computer facility.

Table 3-7.4 Systems No. 1 through No. 3 employing large cards, magnesium construction and containers and full functions performed. No servos are employed; d-c inputs are assumed for pitch and yaw, binary parallel for sine and cosine of roll. This choice of inputs is referred to as optimum inputs; that is, resulting in least complexity.

Table 3-7.5 Systems No. 1 through No. 3 employing small cards, (3.25 x 4.5 x 1 inches), aluminum construction and containers, full functions performed. Three servos are employed to follow-up on three-wire inputs for pitch, yaw and roll.

Table 3-7.6 Systems No. 1 through No. 3 employing large cards, (4.5 x 5.5 x .8 inches) magnesium construction and containers, minimum functions performed. Minimum functions indicates that as many as possible of the computations required have been relegated to an external computer facility, presumably the spacecraft central computer. This should result in the smallest and lightest possible system. No servos are employed.

## TABLE 3-7.3

### SUMMARY SHEET, SYSTEMS WITH SERVOS

TYPE OF SYSTEMS (LARGE CARDS, MAGNESIUM CONSTRUCTION MAGNESIUM CONTAINER, FULL FUNCTIONS PERFORMED)

| System | No. of Servos | No. of Cards | Total Volume Excluding Container (cu. in.) | Total Weight Excluding Container (lbs.) | Container Req'd | Total Weight With Container (Without Shocks) (lbs.) | Total Weight With Container & Shock-Mounts (lbs.) | Power Req'd (Volts Amps) | Reliability (Failure Rate: %/1000 Hrs) | Remarks |
|---|---|---|---|---|---|---|---|---|---|---|
| System No. 1 | 2 | 1 | 105.9 | 3.3 | 2.7 lbs. | 6.1 | 7.1 | 30 | 5.3 | 0.8 Power Factor Lag; 115 VAC,400~, 1∅ |
| System No. 2 Electronics Box | 3 | 23 large | (643.8) 603.8 | (25.2) 23.9 | Long 3/4 ATR | (31.2) 29.9 | (33.2) 31.9 | - | 97.8 | |
| Sighthead | - | - | (242.9) 177.0 | (5.0) 3.0 | 10 lbs. - | (15.0) 13.0 | - | (330) | - | |
| Totals | | | | | | | | 300 | 97.8 | 0.7 Power Factor Lag; 115 VAC, 400~, 1∅ |
| System No. 3 Landing Area Generator | - | large 12 | 239.5 | 8.75 | Short 1/4 ATR | 11.2 | 12.7 | 30 | 45.0 | 0.8 Power Factor Lag; 115 VAC, 400~, 1∅ |
| System No. 3 Total (Landing Area + System No. 2) | | 35 large | | | | | | (360) 330 | 142.8 | 0.7 Power Factor Lag; 115 VAC, 400~, 1∅ |

NOTES:
1. Optics not included
2. Above figures apply to microelectronics versions
3. Large cards, 4.5 x 5.5 x 0.8 inches = Two of small cards 1 x 4.5 x 3.25 inches
4. CRT and yoke reliability figures are lumped in with Electronics Box
5. Figures in parentheses are for two-color display.

TABLE 3-7.4

OPTIMUM INPUTS, SUMMARY SHEET

TYPE OF SYSTEMS (LARGE CARDS, MAGNESIUM CONSTRUCTION, MAGNESIUM CONTAINER, FULL FUNCTIONS PERFORMED)

D-C INPUT FOR PITCH, YAW AND BINARY PARALLEL FOR SIN AND COS OF ROLL

| System | No. of Servos | No. of Cards | Total Volume Excluding Container (cu. in.) | Total Weight Excluding Container (lbs.) | Container Req'd | Total Weight With Container (Without Shocks) (lbs.) | Total Weight With Container & Shock-Mounts (lbs.) | Power Req'd (Volts Amps) | Reliability (Failure Rate: %/1000 Hrs) | Remarks |
|---|---|---|---|---|---|---|---|---|---|---|
| System No. 1 | 2 | 1 | 105.9 | 3.3 | 2.7 lbs. | 6.0 | 7.0 | 30 | 5.3 | 0.8 Power Factor Lag; 115 VAC, 400~, 1∅ |
| System No. 2 | | 27 | 562.8 | 23.5 | Long | 29.5 | 31.5 | - | 108.3 | |
| Electronics Box | - | large | (602.8) | (24.9) | 3/4 ATR | (30.9) | (32.9) | | | |
| Sighthead | - | - | (242.9) 177.0 | (5.0) 3.0 | 10 lbs. - | (15.0) 13.0 | - | - | - | |
| System No. 2 Totals | | | | | | | | (300) 270 | 108.3 | 0.7 Power Factor Lag; 115 VAC, 400~, 1∅ |
| System No. 3 | | 12 | | | Short | | | | | 0.8 Power Factor Lag; 115 VAC, 400~, 1∅ |
| Landing Area Generator | | large | 239.5 | 8.7 | 1/4 ATR | 11.2 | 12.7 | 30 | 45.0 | 0.7 Power Factor Lag; 115 VAC, 400~, 1∅ |
| System No. 3 Totals (Landing Area + System No. 2) | | | | | | | | (330) 300 | 153.3 | 0.7 Power Factor Lag; 115 VAC, 400~, 1∅ |

NOTES:
1. Optics not included
2. Above figures apply to microelectronics versions
3. Large cards 4.5 x 5.5 x .8 inches = Two of small cards 1 x 4.5 x 3.25 inch
4. CRT and yoke reliability figures are lumped in with Electronics Box
5. Figures in parentheses are for two-color display.

TABLE 3-7.5

SUMMARY SHEET, SYSTEMS WITH SERVOS

TYPE OF SYSTEMS (SMALL CARDS, ALUMINUM CONSTRUCTION, ALUMINUM CONTAINER, FULL FUNCTIONS PERFORMED)

| System | No. of Servos | No. of Cards | Total Volume Excluding Container (cu. in.) | Total Weight Excluding Container (lbs.) | Container Req'd | Total Weight With Container (Without Shocks) (lbs.) | Total Weight With Container & Shock-Mounts (lbs.) | Power Req'd (Volts Amps) | Reliability (Failure Rate: %/1000 Hrs) | Remarks |
|---|---|---|---|---|---|---|---|---|---|---|
| System No. 1 | 2 | 1 | 105.9 | 3.3 | 2.7 lbs. | 6.0 | 7.0 | 30 | 5.3 | 0.8 Power Factor Lag; 115 VAC, 400~, 1∅ |
| System No. 2 Electronics Box | 3 | 46 small | 833.8 (873.8) | 28.3 (29.6) | 1 full ATR (Long) | 37.3 (38.6) | 40.3 (41.6) | - | 97.8 | |
| Sighthead | - | - | (242.9) 177.0 | (5.0) 3.0 | 10 lbs. - | (15.0) 13.0 | - | - | - | |
| System No. 2 Totals | | | | | | | | (330) 300 | 97.8 | 0.7 Power Factor Lag; 115 VAC, 400~, 1∅ |
| System No. 3 Landing Area Generator | | 24 small | 354.5 | 11.0 | 1/2 ATR Long | 15.0 | 17.0 | 30 | 45.0 | 0.8 Power Factor Lag; 115 VAC, 400~, 1∅ |
| System No. 3 Totals (Landing Area + System No. 2) | | | | | | | | (360) 330 | 142.8 | 0.7 Power Factor Lag; 115 VAC, 400~, 1∅ |

NOTES:
1. Optics not included
2. Above figures apply to microelectronics versions
3. Large cards, 4.5 x 5.5 x .8 inches = two of small cards, 1 x 4.5 x 3.25 inches
4. CRT and yoke reliability figures are lumped in with Electronics Box
5. Figures in parentheses are for two-color display.

TABLE 3-7.6

SMALLEST POSSIBLE SYSTEM SUMMARY SHEET

TYPE OF SYSTEMS (LARGE CARDS, MAGNESIUM CONSTRUCTION, MAGNESIUM CONTAINER, MINIMUM FUNCTIONS PERFORMED NO SERVOS)

| System | No. of Servos | No. of Cards | Total Volume Excluding Container (cu. in.) | Total Weight Excluding Container (lbs.) | Container Req'd | Total Weight With Container (Without Shocks) (lbs.) | Total Weight With Container & Shock-Mounts (lbs.) | Power Req'd (Volt Amps) | Reliability (Failure Rate: %/1000 Hrs) | Remarks |
|---|---|---|---|---|---|---|---|---|---|---|
| System No. 1 | 2 | 1 | 105.9 | 3.3 | 2.7 lbs. | 6.0 | 7.0 | 30 | 5.3 | 0.8 Power Factor Lag; 115 VAC, 400~, 1ø |
| System No. 2 Electronics Box | - | 14 | 307.0 (347.0) | 10.7 (12.1) | 1/2 ATR Short | 13.5 (14.8) | 15.0 (16.3) | - | 71.9 | |
| Sighthead | - | - | (242.9) 177.0 | (5.0) 3.0 | 10 lbs. - | (15.0) 13.0 | - | - | - | |
| System No. 2 Totals | | | | | | | | 200 | 71.9 | 0.7 Power Factor Lag; 115 VAC, 400~, 1ø |
| System No. 3 Landing Area Generator | - | 4 | 79.0 | 3.0 | Fits into Electronics Box | 3.0 | 3.0 | | 20.3 | 0.8 Power Factor Lag; 115 VAC, 400~, 1ø |
| System No. 3 Totals (Landing Area + System No. 2) | | | | | | | | 200 | 92.2 | 0.7 Power Factor Lag; 115 VAC, 400~, 1ø |

NOTES:
1. Optics not included
2. Above figures apply to microelectronics versions
3. Large cards, 4.5 x 5.5 x 0.8 inches = two of small cards, 1 x 4.5 x 3.25 inches
4. CRT and yoke reliability figures are lumped in with Electronics Box
5. Figures in parentheses are for two color display
6. Use of spacecraft central computer assumed.

The purpose of these four tables is to give a measure of size, weight and power requirements for Systems No. 1 through No. 3 when the conditions described in each of the tables are imposed. Each offers certain desirable features that place it, in some respects, at an advantage as compared to the others. The compromises differ and each must be considered individually. In general, however, the object has been to attain:

1. Maximum package density

2. As high a reliability as possible

3. Minimum weight

4. Least compromise of system capabilities

Certain rules-of-thumb have been employed to facilitate the analysis. It has been assumed that two of the small cards (3.25 x 4.5 x 1 inch) contain the same quantity of electronics as one large card (4.5 x 5.5 x 0.8 inch). In addition, the large card has been assumed to save two ounces of weight if made in aluminum and three ounces if made in magnesium. Along with the larger cards, it is assumed that the total number of connectors is reduced to one-half. This effects another net savings in weight of the order of one ounce per large card. Weights with and without shock mounts are given; need for these has yet to be established. The figures in the four tables do not include the size and weight of the optics but do include the weight of a 3" Dia CRT.

## B. RELIABILITY CONSIDERATIONS

Reliability, like other performance parameters is designed into the system. To insure that a reliable system is being developed, consideration is given to environmental and electrical stresses as well as choice of components in the breadboard or design investigation phase.

The circuits are designed to withstand the maximum environmental stresses to which the system is exposed. To insure that this design objective is reached, all circuits are performance tested at the environmental extremes. Electrical stress levels are calculated and then measured to insure that components will operate within their specifications. To insure maximum reliability, the component stress levels are maintained at a maximum of 50 percent of the maximum specified value. High power components are provided with adequate heat sinks to insure operation within the maximum specified limits.

Many of the circuits used in the systems have been designed using discrete components. These circuits are presently operating in Heads-Up Display equipment being built by Sperry on Bureau of Naval Weapons Contract NOw 63-0190-4. The circuits have been tested over the environmental extremes,

and the stress levels have been maintained at the 50-percent levels specified. Conversion of these circuits to thin film type assemblies, or encapsulated welded cordwood assemblies, will increase the reliability of the system over the use of discrete components soldered on assembly.

The components used in these circuits are high reliability MIL-type components to insure maximum reliability. Use of thin-film substrates would require extensive testing to determine whether they can meet the environmental and electrical stresses to which they will be subjected prior to use in the system. Sperry is conducting such tests on microelectronic integrated circuits on the AN/ARN-76 Loran C receiver. Under this program Sperry is investigating various microelectronic integrated circuits to determine conformity to published specifications under all environmental extremes. Sperry is also negotiating with the Bureau of Naval Weapons to build a microelectronic waveform generator for the Heads-up Display. The information derived from these programs has been used as a basis for the data included in this report.

The circuits used, with the exception of the microelectronic circuits, are packaged in small subassemblies. These subassemblies contain a complete circuit function on either a thin-film substrate assembly or a welded cordwood assembly. The subassembly are then inserted into a printed circuit board assembly where interconnections with other circuits is performed. The printed circuits are held down in the rack by its connector and two hold-down screws.

Transient protection for the system, which also provides further protection for the system components, is provided in the power supplied to meet the applicable requirements of MIL-STD-704.

Reliability figures (MTBF) have been derived for each of the systems under consideration. The results are included in tables 3-7.3 through 3-7.6 and, as may be expected, the reliability varies inversely with system complexity.

The failure rates from the MIL-217 handbook are employed where applicable in the anticipated environments. The microelectronics failure rates are projected figures by Texas Instruments Co., which is presently amassing data of this sort. The Texas Instruments information has been used because there is little else available at this time. The MIL-217 handbook does not, as yet,

contain microelectronics data; however, its information, when applicable, is time proven and more dependable than the Texas Instruments projections.

The results of the reliability analyses are given in Appendix E. The total failure rates for systems No. 1 through No. 3 are presented and are based on using as much microelectronics as possible. Ultimately, the microelectronics package will offer smaller size, weight, power consumption and improved reliability over the conventional component counterparts. However, the systems cannot be made completely microelectronic because certain circuits do not lend themselves to microcircuit implementation. For example, circuits requiring precision resistors, large capacitors, inductors and high power dissipation, cannot be readily replaced by microelectronics.

Redundancy has been considered as a manner of improving reliability; however, it can only be employed to offset outstanding offenders in failure rates.

As shown in Appendix E, redundancy techniques are particularly effective, where required, for specific portions of a system. The disadvantage is that as the number of parts increases, the volume and weight also increase.

In all versions of systems 1 through 3 as described in this report, the failure rates are fairly uniform. There is no element that is a leading contributor to the total failure rate. As a result, redundancy would have to be employed on major subassemblies, such as the Electronics Box or Landing Area Generator. This infers duplication of the entire unit - a high price to pay. Further, it is assumed that detection of a malfunction can be done easily and the standby unit can be substituted into the system with a minimum of difficulty. Further insight on redundancy may be gained from Appendix E where the merits are discussed and qualified.

## C. CATHODE RAY TUBE TRADE-OFF CONSIDERATIONS

The device that involves the largest number of tradeoffs in its selection is the CRT.

### 1. Ultor Voltage

The ultor voltage is subject to compromise. The brightness is directly proportional to the voltage supplied. According to studies conducted by A. Pfahal of Bell Telephone Labs. Inc., brightness can also be raised by increasing the beam current; however, this reduces the phosphor life.

The ultor voltage that the CRT may be operated at is a function of the neck diameter.  A 7/8-inch diameter neck cannot safely go above 15,000 volts; a 1-1/2-inch diameter allows approximately 30,000 volts.  The recommended value used in the present Sperry systems is 25,000 volts; the cost of the higher voltage and brightness is the added power that the deflection circuitry must produce.

Cabling does not impose limits.  The cabling for the 25,000 volts is rated at 40,000 volts at 70,000 feet altitude; similarly, the cabling for the focusing voltages (of the order of several thousand volts) is rated at 15,000 volts.  These high voltage assemblies are shielded for additional safety to personnel.  The cabling is satisfactory for the proposed spacecraft applications described in this study.

## 2. X-Ray Radiation

The X-ray (or gamma effect) radiation has been investigated on a 25,000-volt CRT used presently by Sperry.  Measurements conducted 2 inches from the face of the CRT with no shields or other attenuating appurtenances have found negligible radiation.  This signifies that the level is below that set for Sperry, which is a limit of 80 milliroentgen per 40-hour week or 2 milliroentgen per hour.  This is slightly tighter than the safe level set by the Atomic Energy Commission, 100 milliroentgen per 40-hour week or 2-1/2 milliroentgen per hour.

## 3. Beam Current

A low value of beam current offers another benefit in addition to extended life: the resolution, or spot size, can be reduced.  The beam current and the size of the spot are directly proportional.  The brightness then improves, because it is directly related to beam current per unit area.

## 4. Phosphor

The compromise of attributes extends into the choice of phosphor for the CRT.  Each phosphor has characteristics that distinguish it from the others.  Quite often the choice of certain features is extremely limited and is linked to other characteristics that may or may not be desirable.  To illustrate, each phosphor has fluorescent color, phosphorescent color, spectral peaks, persistence, aging effect and so forth.  Choice of phosphor by one of these characteristics, by necessity, results in inheriting the others. Considering all qualities, a P-1 metallized phosphor is recommended.  This has been proven to be one of the most durable phosphors.  Its efficiency (lumens/watt) is good and its green color offers advantages of its own.  Most high brightness backgrounds are expected to be blue-white in nature, being

sunlight reflections from sky, clouds, sand and water.  Green, the color of
P-1, offers great contrast against these; the images are thus more pronounced
Green is also the color for which the eye has the highest sensitivity.  In
backgrounds of vegetation, a lower light level is anticipated.  In this instance,
the brightness of the cues is the feature that makes them distinguishable.
In a display viewed against the black space background, green, as well as
almost any other color, is suitable.  However, image brightness control
must be used to prevent obscuration of stars  and other less visible objects.
In lunar landing operations, the low reflectivity and expected dark color of
the lunar surface pose no problems with the use of a green phosphor.

The persistance of the metallized P-1 phosphor is acceptable,
being of the order of one image frame time (1/50 second), and thus will not
cause smearing during rapid dynamic operation.

### 5.  Brightness

Assuming that the background brightness can rise to 10, 000
foot-lamberts (5000 to $5500^O$ K color temperature), the estimated required
cue brightness equals 900 foot-lamberts for adequate contrast.  Considering
a reflection coefficient of the combining glass of 20 percent maximum and
an optics efficiency of 60 percent, the system efficiency of CRT screen to
pilot eye is:

$$(0.20) \times (0.60) = 0.12 \text{ or } 12\%$$

Consequently, the brightnesses of the images on the CRT must be:

$$\frac{900 \text{ foot-lamberts}}{0.12} = 7500 \text{ foot-lamberts}$$

This is attainable with the metallized P-1 phosphor.

### 6.  Resolution and Accuracy

Tradeoffs also exist in the resolution and accuracy aspects of
the CRT.  One area affected is the spot size as a function of deflection and
focusing schemes versus the required power.  The CRT can be designed to
have either electrostatic or magnetic deflection and electrostatic and
magnetic deflection focusing.  Electrostatic and magnetic deflection systems
utilize approximately the same amount of power so neither possesses any
significant advantage in this regard.  The magnetic deflection scheme, however,
produces a much smaller spot and thus becomes the logical choice.  The
focusing method can then be considered.  The electrostatic focusing requires
almost no power while the magnetic focusing requires a fixed current flowing
continuously through the focus coil.  Further, the regulation required in the
latter current may dictate a constant current supply.  This would add to the
number of components in the system.  Although magnetic focusing would

produce a further decrease in spot size, the benefits thus derived are not so significant as to warrant the increase in power for the focus coil and possible additional circuitry for the associated constant current supply. The conclusion is the recommendation of a CRT with magnetic deflection and electrostatic focusing.

### 7. Yoke and CRT Dimensions

The physical size of the sighthead is dictated in great part by its constituent cathode ray tube and deflection yoke.

The yoke size is a function of the cue brightness required. As explained earlier, the brighter the images, the higher the ultor voltage required. This, in turn, produces a stiffer electron beam in the CRT. The stiffer beam demands a stronger magnetic field for deflection which is synonymous to a physically larger yoke.

The size of the CRT is a result of compromises in field of view, resolution, and brightness. The field of view and resolution influence the screen diameter directly, while high brightness dictates high ultor voltage and, as a result, a large CRT neck diameter.

The figures in the previous tables in this section are based on a CRT with a 3-inch diameter, a 1-1/2-inch neck, and a length of approximately 9 inches. The associated yoke is 2-3/16 x 4 (dia. ) inches. This combination has been used successfully in display systems similar to those applicable to NASA requirements. However, table 3-7.7 shows variations of CRT nominal face diameter with actual working area, weight and volume to permit size and weight estimates for other CRT's. The optical trade-off charts define the CRT diameter requirements from an optical application standpoint (refer to paragraph 3-6).

CRT's smaller than the 2-inch diameter unit shown in table 3-7.7 are manufactured. These have a smaller neck diameter and are limited to lower ultor voltages (lower brightness for equivalent phosphor life). For missions not requiring the viewing of a heads-up display against a bright (10, 000 foot-lambert) background, such small tubes may be applicable in installations utilizing small exit pupils (e. g. , a helmet-mounted display). A 0.9-inch diameter tube of this type manufactured by Westinghouse is 7-3/8-inches long with a 0.6-inch usable screen diameter. Maximum ultor voltages is 10, 000 volts.

TABLE 3-7.7

TYPICAL CATHODE RAY TUBE SIZES AND WEIGHTS

| Size CRT (diameter in inches) | Weight (lb.) | Maximum Length (inches) | Minimum Usable Screen Diameter (inches) | Representative Dumont Tube | Remarks |
|---|---|---|---|---|---|
| 2 | 0.50 | 7-5/8 | 1-5/8 | 2AP1 | Smallest produced by Dumont |
| 3 | 0.75 | 9-1/8 | 2-5/8 | KC2408P1 | Designed to Sperry spec. |
| 5 | 2.0 | 10-3/4 | 4-1/4 | 5CKP1 | |
| 7 | 4.0 | 13-1/2 | 6 | 7ABP1 | |
| 10 | 9.0 | 17-1/8 | 9 | 10WP1 | |
| 12 | 12.0 | 18-3/8 | 11 | 12ABP1 | |

NOTES:  1.  Diameters are available beyond 12 inches; sizes, however, are prohibitive.
2.  Dumont is presently developing a 1-inch diameter tube; detailed information not available at present.
3.  All indicated tubes feature magnetic deflection and electrostatic focusing. Yoke is the same for all:  Syntronic Instruments Inc., Type, Y-64, 2-3/16 x 4.0 inches diameter, and weight is 1.25 lbs.

# SECTION IV

## CONCLUDING REMARKS AND RECOMMENDATIONS

### 4-1. CONCLUDING REMARKS

The objectives of the current study program have been to furnish NASA with information which will permit the determination of the application of projection display techniques to spacecraft, and to provide information which will permit NASA to establish a complete simulation program to evaluate these display techniques.

These studies have resulted in the generation of a design specification which defines a display system that may be used in a simulation program. Design investigations have analyzed information requirements for manned spacecraft control during rendezvous and lunar landing. These information requirements, along with specific application criteria, have been used to establish display configurations for the two missions utilizing three levels of display system comprehensiveness.

Trade-off studies have covered both electronic and optical equipment aspects and charts have been furnished summarizing these studies for use in specific applications. The integration of projection displays into a cockpit, considering the corresponding panel display requirements and modifications, has been investigated. The compatibility of display and controls for the rendezvous and lunar landing missions has also been covered.

The results of these studies as presented in this report have therefore satisfied the program objectives.

### 4-2. RECOMMENDATION

The key to the use of projection display techniques in manned space flight is the demonstration of improved flight control performance for critical mission phases. As part of NASA's consideration for a simulation program or for further studies, the following recommendations are made:

1. The manned flight control performance improvement with projection displays for rendezvous and lunar landing should be demonstrated in a simulation program.

2. Further analytical and simulation studies of path and speed control techniques for lunar landing should be conducted as a basis for the NASA simulation program.

3. The modified 2-impulse rendezvous method, which suggests a simplified flight handbook approach to this maneuver with minimum system complexity, should be investigated in more detail.

4. A whole-system trade-off analysis of the use of projection displays in a specific spacecraft type should be conducted. The trade-off should cover projection displays in the window, use of dual overlapping monoculars, and current spacecraft instrumentation. Trade-off parameters should include total spacecraft weight and power increments. Structural weight aspects as window requirements change, and power requirements, including changes in panel illumination requirements, for example, should be considered. Total sensor requirements and accuracies should be covered.

## LIST OF REFERENCES

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2. Final Engineering Report, Development of Advanced Approach and Landing Display, furnished by Sperry Gyroscope Company to Aeronautical System Division, AF System Command, USAF, Wright-Patterson AFB, Ohio, under Cohtract No. AF 33(616)7690.

3. Final Engineering Report, Development of Visual Landing Display, furnished by Sperry Gyroscdpe Company to Avionics Division, Bureau of Naval Weapons, Department of Navy, Washington, D. C.

4. System Analysis Report on Heads-Up Display System, furnished by Sperry Gyroscope Company to Avionics Division, Bureau of Naval Weapons, Washington, D. C., under Contract No. NOw 63-0190-d.

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6. Richard A. Hord, Relative Motion in the Terminal Phase of Interception of a Satellite or a Ballistic Missile, NASA TN-4399, September 1958.

7. W. G. Green, "Logarithmic Navigation for Precise Guidance of Space Vehicles", PGANE - IRE, June 1961.

8. Basil Staros, "Rendezvous in Orbit", Sperry Engineering Review, Pages 2-8, Summer 1962.

# APPENDIX A

## DESIGN SPECIFICATION FOR AN ANALOG SIMULATOR
## DISPLAY SYSTEM (SPERRY DOCUMENT 9202-1002)


A copy of the Design Specification for an Analog Simulator Display System is presented on the following pages.

Design Specification for an Analog Simulator Display System


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| CODE | | | | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| SPECIFICATION NO. | | 9 | 2 | 0 | 2 | – | 1 | 0 | 0 | 2 | | | | |
| LATEST REVISION | | | | | | | | | | | | | | |
| SUB-LETTER | | | | | | | | | | | | | | |

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## 1.0    Scope

## 1.1    Introduction:

This specification defines the technical requirements for a display system for use in a fixed base spacecraft analog simulator in accordance with the requirements of Work Statement 1.03A of contract NAS 4-385, Integrated Spacecraft Display Development.

The basic display system is adaptable for use with a variety of spacecraft simulators. This specification, however, defines the requirements of a projection display system for use with NASA simulators which use analog computers, types 31R, 131R, and 231R.

## 1.2    General Description:

The display system will be used to present to the astronaut various forms of flight control and assessment information on a window or windshield so that the information is superimposed on the simulated real world display of the fixed base spacecraft analog simulator.

The display information, where desirable, will be compatible on a one-to-one relationship with the simulated real world display normally utilized by the pilot in the spacecraft simulator. The display information shall consist of artificially generated images which appear to emanate from infinity in the pilot's forward field of view. The images will be generated from cathode ray tube display and from illuminated reticle information and will pass through an optical system which will collimate the light rays.

The display system will have a suitable capacity and flexibility of display information as specified herein for experimental evaluation of same in the spacecraft simulator.

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2.0     Requirements:
        The system shall meet all requirements specified below.

2.1     Operation:
        The equipment shall provide the type of display information,
herein specified, when used in a fixed base spacecraft simulator with
analog computers of the 31R, 131R, and 231R types.
        The equipment shall be usable, with some modification, in installa-
tions of the overhead—mounted or of the panel—mounted Sighthead type.


2.2     Field of View (FOV)


2.2.1   Total Field of View:
        The CRT display information shall cover a minimum field of view of
20 degrees diameter.
        Provision shall be provided to laterally shift this field of view a
minimum of $\pm 10$ degrees about the spacecraft yaw      axis by means
of an electromechanical servo.  The field of view shall always be
laterally centered around the reticle—generated cue (Form 2).


2.2.2   Instantaneous Field of View:
        The instantaneous field of view shall be as follows for an eye to
lens distance of 22 inches and with bifocal vision:
        Without head motion, the FOV shall be 14 degrees vertical and
20 degrees lateral.
        The full 40—degree lateral FOV (i.e. the 20—degree field plus the
additional $\pm 10$ degree shift) shall be viewable with a maximum lateral
head motion of $\pm 2$ inches.
        It is recommended that installations be arranged so that the eye to
lens distance does not exceed a maximum of 24 inches.

2.2.3   Optical Boresight Axis Reference
        The boresight axis of the optical field of view shall be within
$\pm 10°$ of the spacecraft fuselage reference axis.



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2.3       **Display**

2.3.1     **System Capacity:**
          The equipment shall have capacity to generate and display,
          through suitable mode selection, the cues specified in 2.3.3, Cue
          Forms.
          Provision shall be included to allow the cues to be changed in
          size and shape, with some modification to the equipment, as required,
          during the course of the program.

2.3.2     **Cue Characteristics**

2.3.2.1   **Cue Size:**
          All cues are of fixed preset size and shape unless otherwise noted.

2.3.2.2   **Cue Source:**
          All cues are generated by the CRT only except those specified as
          being also generated by the reticle.

2.3.2.3   **Cue Color:**
          All reticle-generated cues are of red color.  All CRT-generated
          cues are of green color.  The equipment shall have provision for the
          addition of filters so all cues are red for night adaptation.

2.3.2.4   **Cue Position**

2.3.2.4.1 **Normal Operation:**
          For normal operation, the cues shall be positioned and used for the
          functions specified in 2.3.3, Cue Forms, and 2.3.4, Input Signal
          Requirements.

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All cue positions will be attitude stabilized by the Analog Simulator Display System computer unless otherwise specified. Cues, specified as non-attitude stabilized, will require proper signals from the fixed base simulator computers if attitude stabilization is desired.

2.3.2.4.2 <u>Experimental Operation</u>:

For experimental purposes, the cues may be used for other than the normal function. System switching will allow each cue position to be directly controlled from external signals as required.

The cue position will remain attitude stabilized through internal system computation unless otherwise specified.

2.3.2.5 <u>Cue Position Range</u>:

Cues generated by the CRT shall have a vertical and lateral position range equal to the vertical and lateral FOV specified in 2.2.

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2.3.3     Cue Forms

2.3.3.1   Cue Form 1 Fuselage Reference (Figure I-1):
          This cue shall provide a fuselage reference bore sight image which
is fixed at the intersection of the spacecraft's longitudinal and
lateral planes.
          Normal use for this cue shall be as a fixed reference for the
determination of attitude.
          The cue may be generated by the CRT or the reticle.


2.3.3.2   Cue Form 2 Flight Path Marker or Director (Figure I-2):
          Normal use for this cue shall be as a moving indication of space-
craft velocity direction or for use as director information.
          The "wings" shall remain parallel to the spacecraft's lateral axis
at all times.
          The cue may be generated by the CRT or the reticle.


2.3.3.3   Cue Form 3 Symbolic Error (Figure I-3)
          This cue may be normally used for thrust, acceleration or velocity
error indication.  The error is displayed by the position of the bar
relative to the "wing tip" of Cue Form 1 or 2.

          Cue Form 3A
          This cue is CRT generated and may be used with the CRT-generated
forms of Cue 1 and Cue 2.

          Cue Form 3B
          This cue form is identical to 3A but is generated by the reticle
source, and, as such, is usable only with the reticle-generated forms
of Cue 1 and Cue 2.

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2.3.3.4   Cue Form 4   Command Attitude (Figure I-4)
          Cue Form 4A   Pitch-Yaw Command
              Normal use for this cue shall be to display command attitude
          (Pitch and Yaw ).  Control is accomplished by changing spacecraft
          attitude until the Fuselage Reference cue overlays the command attitude cue.
              The dashed line indicates the most direct attitude change to
          achieve the command attitude. This cue is not attitude stabilized.


          Cue Form 4B   Roll-Pitch-Yaw Command
              This cue form is identical to 4A with the addition of Command Roll.
          Control is achieved by changing spacecraft attitude until the Fuselage
          Reference cue overlays the Command Attitude cue in pitch, heading and
          roll aspects. This cue is not attitude stabilized.


2.3.3.5   Cue Form 5   Angular Velocity and/or Acceleration, Damping or Predicted
          Information:  (Figure I-5)
              Normal use for these cues shall be for attitude stabilization and/or
          control via angular velocity and/or acceleration damping, predicted or
          director information.
              The cues shall always be positioned relative to the Fuselage
          Reference Cue No. 1 (CRT type).


          Cue Form 5A   Roll Function
              This cue shall display information of the angular rate/acceleration
          about the roll axis of the spacecraft.


          Cue Form 5B   Pitch-Yaw Function
              This cue shall display angular velocity/acceleration information
          about the pitch and yaw axes of the spacecraft.


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2.3.3.6    <u>Cue Form 6</u>  <u>Local Horizon</u>  (Figure I-6)
This cue, when used with the fuselage reference cue No. 1, shall
provide qualitative pitch and roll attitude information.


2.3.3.7    <u>Cue Form 7</u>  <u>Orbital Plane</u> (Figure I-7)
This cue of the orbital plane is normally for attitude reference
use and, in particular, orbital changing operations.


2.3.3.8    <u>Cue Form 8</u>  <u>Orbital Flight Path Angle</u> (Figure I-8)
The position of this cue relative to the local horizon is the
flight path angle of the spacecraft in the orbital plane.
Qualitative flight path information results when cue is read against
the Local Horizon.  Quantitative information results when the cue is
read against the Pitch Scale cue.


2.3.3.9    <u>Cue Form 9</u>  <u>True Horizon</u> (Figure I-9)
This cue provides a reference horizon line which will overlay the
true horizon.
When the spacecraft is in the vicinity of the earth, the true
horizon is the earth horizon.  When the spacecraft is in the vicinity
of the moon, the true horizon is the moon horizon.


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2.3.3.10   <u>Cue Form 10</u>   <u>Ground Track</u>   (Figure I-10)
This cue shall provide an image that will overlay the past and
predicted future ground track of the spacecraft.


2.3.3.11   <u>Cue Form 11</u>   <u>Attitude Index Marks</u> (Figure I-11)

<u>Cue Form 11A</u>
This cue shall provide attitude index marks on the Local Horizon
(Cue 6) at 90 and 270° heading points relative to the orbital plane.

<u>Cue Form 11B</u>
This cue shall provide index marks on the orbital plane (Cue 7)
at the Zenith and Nadir.


2.3.3.12   <u>Cue Form 12</u>   <u>Pitch Attitude Scale</u> (Figure I-12)
Two pitch attitude scales shall be provided with a range of ±40°.
Each scale shall be positioned at the intersection of the Local Horizon
cue with the Orbital Plane cue; one looking in the direction of the
spacecraft velocity direction and the other looking 180° from the
velocity direction.
The scales may be used for pitch, flight path angle and dip angle
measurements.


2.3.3.13   <u>Cue Form 13</u>   <u>All Attitude Pitch Scale</u> (Figure I-13)
This cue shall provide a suitable pitch attitude scale for all-
attitude maneuvers.
The scale shall remain laterally centered on the spacecraft yaw axis.


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2.3.3.14  Cue Form 14  Star Target #1  (Figure I-14)
          A cue of fixed circular form and size shall be provided for use
with star target sighting.
          This cue may be positioned automatically from known data inputs
for star location or from manual inputs for azimuth and elevation
measurements on a particular star in view.


2.3.3.15  Cue Form 15  Star Target #2  (Figure I-15)
          This cue of fixed rectangular form and size shall be used in the
same manner as Cue Form 14.


2.3.3.16  Cue Form 16  Space Object Target  (Figure I-16)
          A cue of fixed circular form and size shall be provided for use
with space object target sighting.
          The cue may be positioned automatically from data inputs (with
known target position information) for target location or from manual
inputs for azimuth and elevation measurements on a particular target
in view.


2.3.3.17  Cue Form 17  Stadiometric Ranging  (Figure I-17)
          A cue of fixed circular form shall be provided for use with stadio-
metric ranging of targets; i.e.:  distance measured along a direct
bearing from spacecraft to target.
          The size of the cue shall vary as a function of distance.


2.3.3.18  Cue Form 18  Ground Target #1  (Figure I-18)
          A cue of fixed circular form and size shall be provided for use
with ground target sighting.
          The cue may be automatically positioned from known data input for
target location or from manual inputs for azimuth and elevation measure-
ments on a particular object in view.

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2.3.3.19   Cue Form 19  Special  (Figure I-19)
          A cue of fixed circular form is provided for control directly by
external signals.  The cue is not attitude stabilized.


2.3.3.20   Cue Form 20  Altitude Error  (Figure I-20)
          The moving bar cue read relative to the horizon cue provides
altitude error information.  A scale of two fixed index reference
marks at ±100 feet altitude is also provided.


2.3.3.21   Cue Form 21  Attitude or Rate Command (Alternate)  (Figure I-21)

          Cue Form 21A  Pitch-Yaw Axes Command
          Proper pitch-yaw attitude or rate control is achieved by
pitching and yawing until the pitch and yaw command bars are
"zeroed" over the fuselage reference cue.  This cue is not attitude
stabilized.

          Cue Form 21B  Roll-Pitch-Yaw Axes Command
          Proper pitch-yaw control is achieved as in A above.  Proper
roll command is achieved when the fuselage reference "wings" are
aligned parallel to the pitch-roll CMD bar.  This cue is not attitude
stabilized.


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2.3.4    Input Signal Requirements:
         The equipment shall position the cues in accordance with the
functions specified in 2.3.3 when provided with input signal
information as specified in Table 1, Table of Input Signal Informa-
tion Requirements.


2.3.5    Mode Selection:
         The equipment shall contain suitable switching for selection of
the cue forms specified in 2.3.3 for display in either the normal mode
(2.3.2.4.1) or the experimental mode (2.3.2.4.2) of operation.
         In addition, controls for the following shall be provided:
                   1)  Power:          Off—Standby—Operate Switch
                   2)  CRT Display:     On—Off Switch
                   3)  CRT Display:     Brightness Control
                   4)  Reticle Display: On—Off Switch
                   5)  Reticle Display: Brightness Control


2.3.6    Power Requirements:
         The system shall utilize the following types and amount of power.

2.3.6.1  Standby Power:
         AC:  400 cps, 1 phase, 115 volts, 200 watts
         DC:  27.5 volts, 10 watts

2.3.6.2  Operating Power:
         AC:  400 cps, 1 phase, 115 volts, 350 watts
         DC:  27.5 volts, 25 watts


2.3.7    Environmental Requirements:
         The system shall meet all requirements specified in section 2.0
when operated under the normal environmental conditions encountered in
use with a fixed base type simulator.


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3.  **Equipment:**
    The system shall consist of the following major units; each of which is separately mounted.
    1) Sighthead Unit
    2) Electronics Unit #1
    3) Electronics Unit #2
    4) High Voltage Power Supply

3.1 **Sighthead:**
    The Sighthead shall be an improved type of the model shown in Figure II. It shall provide an optical channel for CRT–generated images and a second optical channel for reticle–generated images.

    The sighthead shall be usable for both overhead–mounted or panel–mounted installations with some modification of mirrors and lenses. The bore sight axis of the optical field of view shall be within $\pm 10°$ of the spacecraft fuselage reference axis.

3.2 **Electronics Units:**
    Two Electronics units shall contain the electronics for the system, except the High Voltage Power Supply. This includes all pattern generation, timing, blanking, analog computation, coordinate conversion, deflection amplifiers, low voltage power supply circuitry, mode, power and adjustment controls.

    All circuitry shall be contained on plug–in cards.

    The Electronics Unit shall consist of a standard 19–inch wide lab rack with the necessary plug–in circuitry modules.

    The Electronics Units, with some modifications, shall be usable with installations of either the overhead–mounted or the panel–mounted Sighthead.

3.3 **High Voltage Power Supply (HVPS):**
    The HVPS provides the high voltage required for the CRT. The unit shall be sealed and is capable of being hard mounted to the cockpit.

    The HVPS must be mounted within 3 feet of the Sighthead.
    Size and weight of the HVPS shall be as shown in Figure III.

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# Table 1

## Table of Input Signal Information Requirements

1. **Roll Attitude $\phi$**

   a. D.C. linear signal:
      +100V DC = +540°, 0V DC = 0°, −100V DC = −540°

   b. DC Sin and Cos signals:
      $E_1 = 100V \sin \phi$    $E_2 = 100V \cos \phi$

   c. Synchro signal:
      3-wire, 11.8V leg to leg, 400 cps, from synchro transmitter.

   Note: Scale zero reference is zero spacecraft roll attitude as measured relative to local horizon.

2. **Pitch Attitude $\theta$**

   a. DC linear signal:
      +100V DC = +90°, 0V DC = 0°, −100V DC = −90°

   b. DC Sin and Cos signals:
      $E_1 = 100V \sin \theta$, $E_2 = 100V \cos \theta$

   c. Synchro signal:
      3-wire, 11.8 volts leg to leg, 400 cps, from synchro transmitter.

   Note: Scale zero reference is zero spacecraft pitch attitude as measured relative to local horizon.

3. **Yaw Attitude $\psi$**

   a. DC linear signal:
      +100V DC = +360°, 0V DC = 0°, −100V DC = −360°

   b. DC Sin and Cos signals:
      $E_1 = 100V \sin \psi$ , $E_2 = 100V \cos \psi$

   c. Synchro signal:
      3-wire, 11.8V leg to leg, 400 cps, from synchro transmitter.

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Note: Scale zero reference is zero spacecraft yaw attitude as measured relative to the orbital plane in the forward direction of velocity.

4. Command Roll Attitude $\phi c$

   a. DC Sin and Cos signals:
      $E_1 = 100V \sin \phi c$, $E_2 = 100V \cos \phi c$

   Note: Command roll attitude is difference between present and required roll attitudes.

5. Command Pitch Attitude $\theta c$

   a. DC linear signal: $\pm 1V$ DC/Degree

   Note: Command pitch attitude is difference between present and required pitch attitudes.

6. Command Yaw Attitude $\psi c$

   a. DC linear signal: $\pm 1V$ DC/Degree

   Note: Command yaw attitude is difference between present and required yaw attitudes.

7. Roll Rate $\dot{\phi}$

   a. DC linear signal: $\pm 100V$ DC $= \pm$ Full Scale, $0V = 0$ rate

8. Pitch Rate $\dot{\theta}$

   a. DC linear signal: $\pm 100V$ DC $= \pm$ Full Scale, $0V = 0$ rate

9. Yaw Rate $\dot{\psi}$

   a. DC linear signal: $\pm 100V$ DC $= \pm$ Full Scale, $0V = 0$ rate

10. Roll Acceleration $\ddot{\phi}$

   a. DC linear signal: $\pm 100V$ DC $= \pm$ Full Scale, $0V = 0 \ddot{\phi}$

11. Pitch Acceleration $\ddot{\theta}$
   a. DC linear signal: $\pm 100V$ DC $= \pm$ Full Scale, $0V = 0 \ddot{\theta}$

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12. <u>Yaw Acceleration</u> $\ddot{\psi}$

    a. DC linear signal: ±100V DC = ± Full Scale, 0V = 0 $\ddot{\psi}$

13. <u>Orbital Flight Path</u>

    a. DC linear signal: +100V DC = +90°, 0V = 0°, −100V = −90°

       Note: Zero scale reference is the local horizon.

14. <u>Flight Path</u>

    a. DC linear signal: +100V DC = +90°, 0V = 0°, −100V = −90°

       Note: Zero scale reference is the local horizon.

15. <u>Drift Angle</u>

    a. DC linear signal: +100V DC = +90°, 0V = 0°, −100V = −90°

   Note: Zero scale reference is the spacecraft yaw axis.

16. <u>Thrust Error</u>

    a. DC linear signal: ±10V DC=±Full Scale, 0V DC = 0 thrust error

17. <u>Longitudinal Velocity Error</u>

    a. DC linear signal: ±10V DC=±Full Scale, 0V DC = 0 velocity error

18. <u>Dip Angle</u>

    a. DC linear signal: −100V DC = −90°, 0V DC = 0°

       Note: Scale zero reference is local horizon.

19. <u>Ground Track</u> $\psi_T$

    a. DC Sin and Cos signal:
       $E_1 = 100V \, \text{Sin} \, \psi_T$, $E_2 = 100V \, \text{Cos} \, \psi_T$

       Note: Scale zero reference is orbital plane.

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20. **Star Target #1**

   **Azimuth:** DC linear signal: ±100V DC = ± 90°
   Scale zero reference is spacecraft yaw axis.

   **Elevation:** DC linear signal: ±100V DC = ±90°
   Scale zero reference is local horizon.

21. **Star Target #2**

   **Azimuth:** DC linear signal: ±100V DC = ± 90°
   Scale zero reference is spacecraft yaw axis.

   **Elevation:** DC linear signal: ±100V DC = ±90°
   Scale zero reference is local horizon.

22. **Space Object**

   **Azimuth:** DC linear signal: ±100V DC = ± 90°
   Scale zero reference is spacecraft yaw axis.

   **Elevation:** DC linear signal: ±100V DC = ±90°
   Scale zero reference is local horizon.

23. **Stadiometric Ranging Target Info.**

   1) **Range:** DC linear signal: +100V DC=±Full Scale, 0V = 0 dist.

   2) **Azimuth:** DC linear signal: ±100V DC = ±90°
      Scale zero reference is spacecraft yaw axis.

   3) **Elevation:** DC linear signal: ±100V DC = ±90°
      Scale zero reference is local horizon.

24. **Ground Target**

   1) **Azimuth:** DC linear signal: ±100V DC = ±90°
      Scale zero reference is spacecraft yaw axis.

   2) **Elevation:** DC linear signal:−100V DC = −90°
      Scale zero reference is local horizon.

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25. <u>Special Symbol (Cue 19)</u>

    1)  X Position:  DC linear signal:  ±100V DC = ± Full Scale , OV = O X displacement,
                   Scale zero reference is center of display.

    2)  Y Position:  DC linear signal:  ±100V DC = ± Full Scale, OV = O Y displacement
                   Scale zero reference is center of display.

26. <u>DC Reference Signals</u>

    1)  +100V DC

    2)  −100V DC

27. <u>Altitude Error</u>

    a.  DC linear signal:  ±100V DC = ± Full Scale, OV DC = 0 altitude error.

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FIG I-1    CUE FORM 1
FUSELAGE  REFERENCE

CRT TYPE    RETICLE TYPE

FIG I-2    CUE FORM 2
FLIGHT PATH MARKER OR DIRECTOR
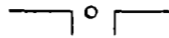
FIG I-3    CUE FORM 3

SYMBOLIC ERROR — THRUST, ACCEL, VELOCITY

3A        3B
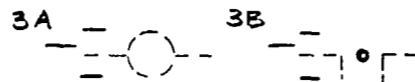
FIG I-4    CUE FORM 4
COMMAND ATTITUDE
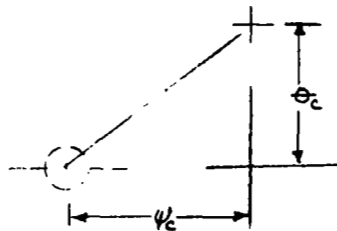
4 A) CMD PITCH & YAW          4B) CMD ROLL PITCH & YAW

$\theta_c$

$\psi_c$

$\psi_c$

$\theta_c$

$\psi_c$
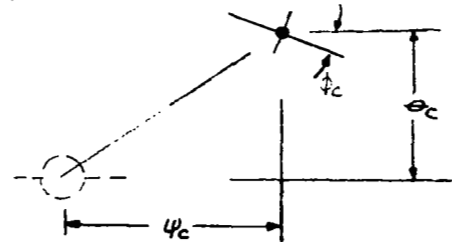
FIGS. I-1, 2, 3, 4

FIG I-5     CUE FORM 5

ANGULAR VELOCITY AND/OR ACCELERATION, DAMPING OR
PREDICTED INFORMATION

5A) ROLL FUNCTION

$$\dot{\phi} = -\tfrac{1}{2}K \qquad \dot{\phi} = 0 \qquad \dot{\phi} = +\tfrac{1}{2}K \qquad \dot{\phi} = +1K$$

5B) PITCH & YAW FUNCTION

$$\dot{\psi} = 0 \qquad \dot{\psi} = 0 \qquad \dot{\psi} = 0 \qquad \dot{\psi} = 0$$
$$\dot{\theta} = +\tfrac{1}{2}K \qquad \dot{\theta} = 0 \qquad \dot{\theta} = -\tfrac{1}{2}K \qquad \dot{\theta} = -1K$$

$$\dot{\psi} = -\tfrac{1}{2}K \qquad \dot{\psi} = +\tfrac{1}{2}K \qquad \dot{\psi} = +1K \qquad \dot{\psi} = +1K$$
$$\dot{\theta} = 0 \qquad \dot{\theta} = 0 \qquad \dot{\theta} = -\tfrac{1}{2}K \qquad \dot{\theta} = -1K$$
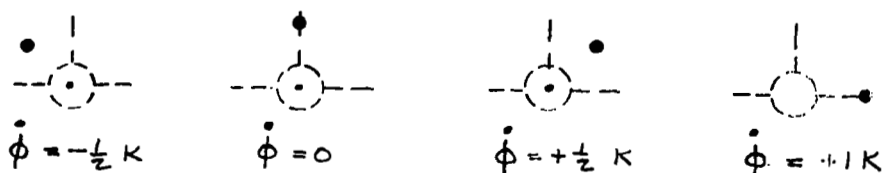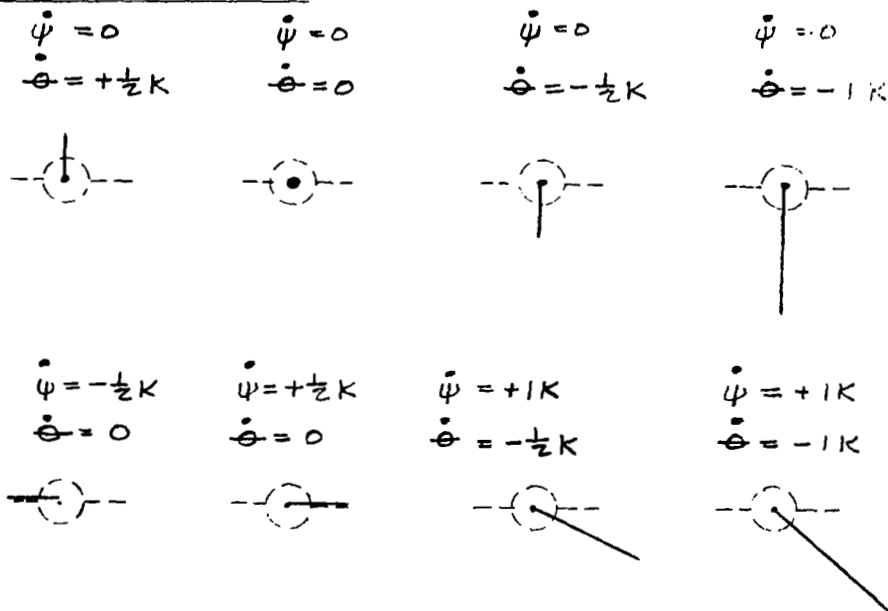
FIGURE I-5

## FIG I-6    CUE FORM 6
## LOCAL HORIZON
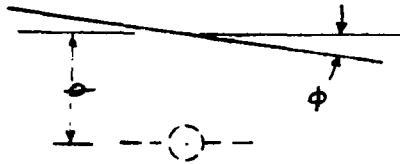


## FIG I-7    CUE FORM 7
## ORBITAL PLANE



ORBITAL
PLANE

LOC HOR
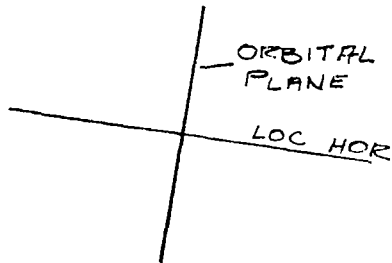
## FIG I-8    CUE FORM 8
## ORBITAL FLIGHT PATH ANGLE $\gamma$

i)  FORWARD VIEW - ALONG        VIEW 180° FROM i)
    VELOCITY DIRECTION



$\gamma$

LOC HOR

ORB PLANE

LOC HOR
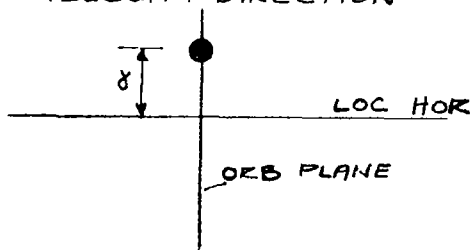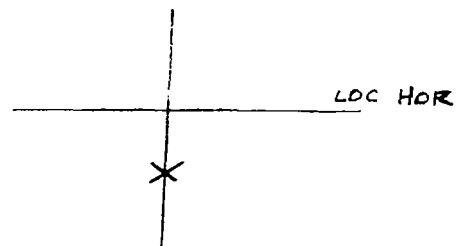
FIGS I-6 7 & 8

FIG I-9   CUE FORM 9
TRUE HORIZON



ORB
PLANE

LOC HOR

DIP ANGLE

x

TRUE HORIZON

FIG I-10   CUE FORM 10
GROUND TRACK



ORB
PLANE

LOC HOR

x

TRUE HOR
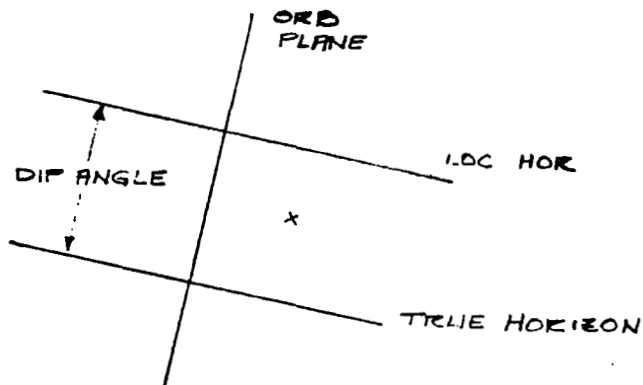
GROUND
TRACK —

FIGURES I-9 & 10

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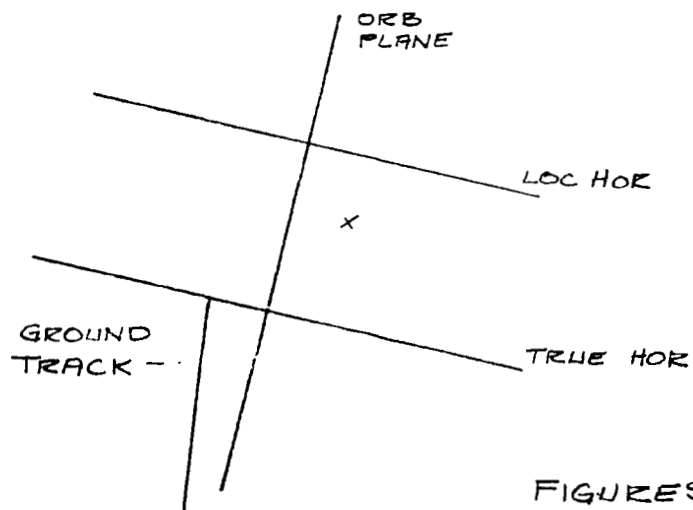
FIG I-11   CUE FORM 11
ATTITUDE INDEX MARKS

11 A)   90° & 270°  YAW INDICES       11 B)   ZENITH & NADIR
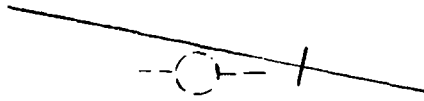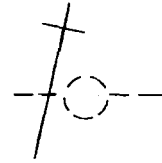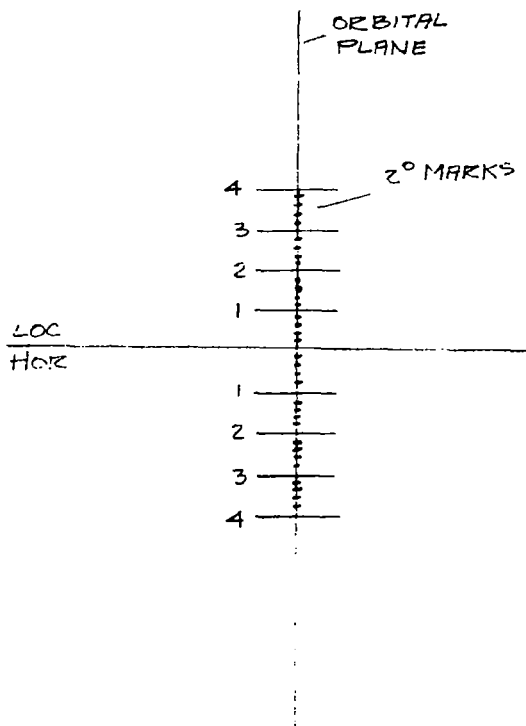


FIG I-12   CUE FORM 12                 FIG I-13  CUE FORM 13
PITCH   SCALE                          ALL ATTITUDE SCALE
                                            ✛  ZENITH



ORBITAL
PLANE

2° MARKS

4
3
2
1

LOC
HOR

1
2
3
4

70
50
30
10
10
30
50
70
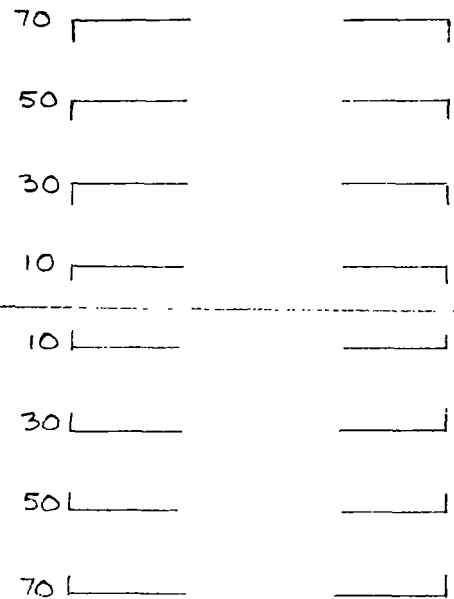
◯ — NADIR

FIGURES  I-11, 12 & 13

FIG. I-14  CUE FORM 14  STAR TARGET #1

FIG I-15  CUE FORM 15  STAR TARGET #2

FIG I-16  CUE FORM 16  SPACE TARGET

FIG I-17  CUE FORM 17  STADIOMETRIC RANGING

FIG I-18  CUE FORM 18  GROUND TARGET

FIG I-19  CUE FORM 19  SPECIAL

FIGS. I-14, 15, 16, 17, 18, 19

FIG I-20     CUE FORM 20

ALTITUDE ERROR

—        +100'
          INDEX

_____↓_____ LOC. HOR

ALT ⌐_____—     −100'
ERR. ⌐↑    —      INDEX
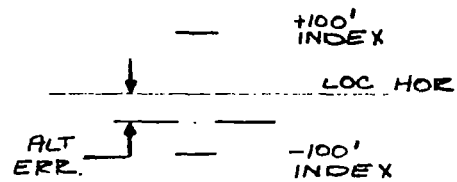

FIG I-21     CUE FORM 21

COMMAND ATTITUDE OR RATE

21 A)  PITCH-YAW CMD      21 B)  ROLL-PITCH-YAW CMD

$\theta_c$

$\psi_c$

$\theta_c$

$\phi_c$

$\psi_c$
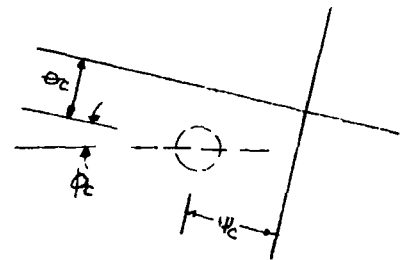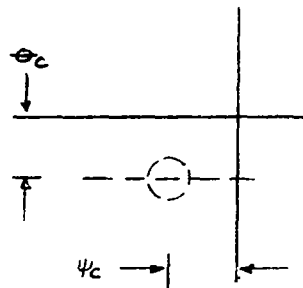

FIGS I-20, 21

# APPENDIX B

## DESCRIPTION OF COCKPIT MOCKUPS

Two typical spacecraft cockpit mockups have been fabricated to permit a realistic application of the results of the trade-off analyses. These mockups are shown in figures B-1 and B-2.

## B-1. DESIGN

Initially, it was thought that a universal mockup could be constructed which would provide a cockpit compatible with the type expected to be used by Apollo, LEM and Gemini. However, the visual field provided to the astronaut in the Apollo and Gemini windows and the expected LEM windows proved to be so different that another concept had to be formulated. One solution to the problem appeared to be the construction of the mockup in three units: a base unit which would contain the seats, pedestal and rear bulkhead; and two other units which would be representative of the Gemini and LEM cockpits, each interchangeable with the base unit. A quarter-scale model was built following this concept.

The model brought out the fact that this approach was not acceptable because of the difficulty in properly locating the astronaut relative to the windows. Information obtained about the LEM indicated that the astronaut would be much closer to this window than in the Gemini. Consequently, a final concept was evolved which would be acceptable for a typical "orbital rendezvous" cockpit and for a typical "lunar landing" cockpit. This concept was to build two separate units which would encompass one-half of a cockpit for each type.

The mockups are to be used as generalized versions of spacecraft cockpits for control/display and panel/windshield display integration. Therefore, construction was not elaborate or detailed. Dimensions were obtained from several sources for both mockups, including trade magazines and, in the case of the LEM, from the prime contractor. Although the mockups are not exact replicas, their physical sizes approximate actual spacecraft dimensions.

## B-2. CONSTRUCTION

One of the mockups was based on the Gemini spacecraft. External dimensions of the Gemini were used as a guideline, including the dimensions of the entrance/egress hatch. The left half of the Gemini was built by bisecting

the spacecraft longitudinally along the spacecraft plane of symmetry between the two astronaut seats. The astronaut's seat in the Gemini vehicle is rotated approximately twelve degrees counterclockwise from the vertical plane of symmetry. For convenience in laboratory use and to avoid a tilted seat, the cockpit base was made to tilt the entire cockpit unit 12 degrees clockwise. To reduce the overall height of the mockup, the plane of the base was located at the astronaut's foot level. Forms were constructed and located with stringers and the external surface was covered with aluminum. A removable hatch was constructed which contains the window framework. An instrument panel bulkhead, an overhead panel bulkhead, and a center pedestal are in place for use as support structures for instrument and control mockups. The center pedestal has been hinged to allow access to the seat, and a dummy controller has been placed on the center pedestal. The exterior has been painted white and the interior a flat gray.

The LEM mockup was constructed with preliminary information received from the LEM contractor. Construction had been started based on early information which indicated that a tiltable seat was included to increase the astronauts field of view during the critical landing phase. After constructing a seat which would achieve the desired tilt angle, LEM design changes eliminated the seat and a completely new window and forward cockpit area were designed. Since the tiltable seat in the mockup was already constructed, it has been retained. However, the frontal area and window configuration were redesigned to conform to the current LEM configuration. The shell which encloses the left half of the cockpit is covered with aluminum. The exterior is white and the interior a flat gray, similar to the Gemini mockup.
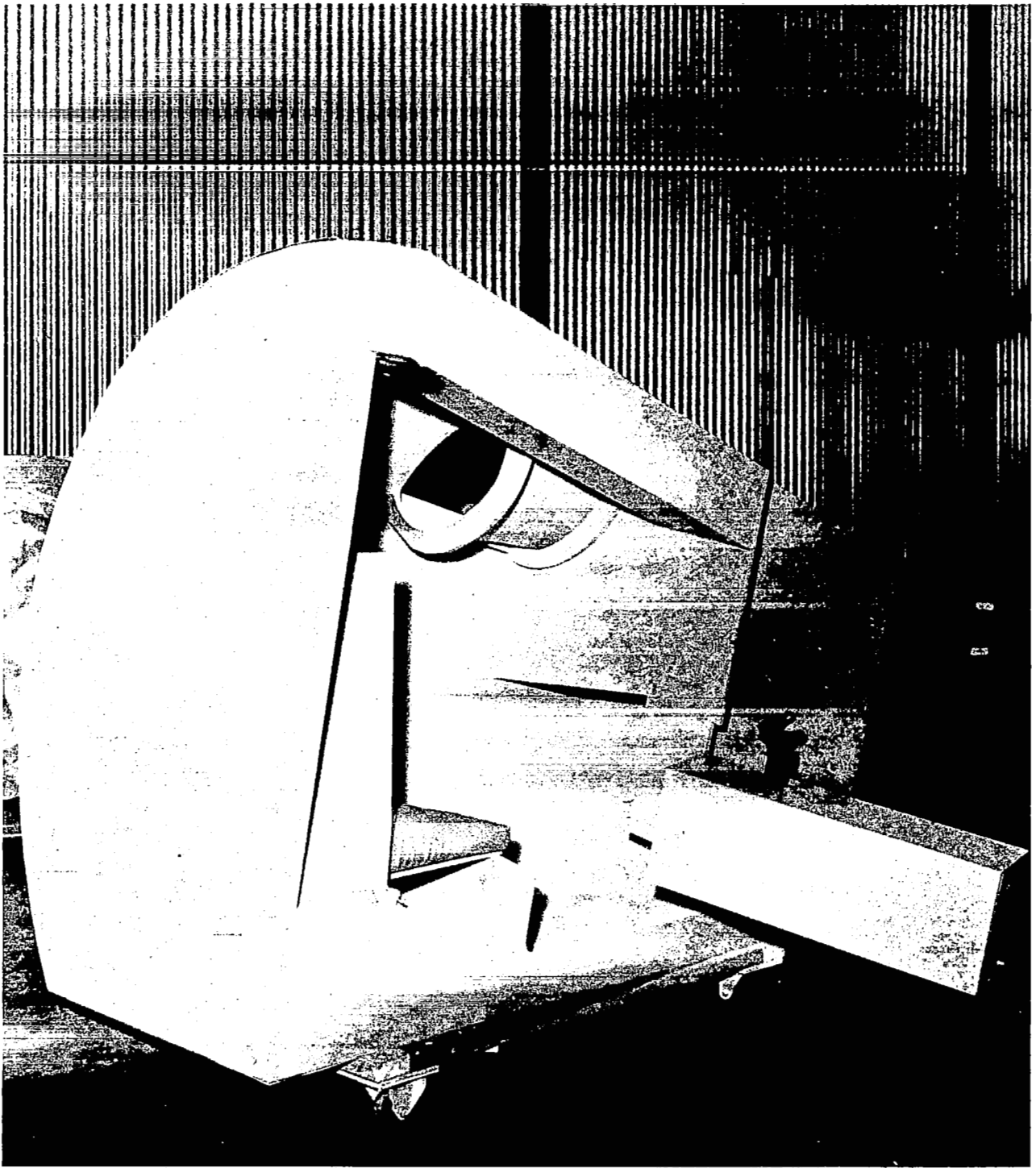
FIGURE B-1 GEMINI COCKPIT MOCK UP
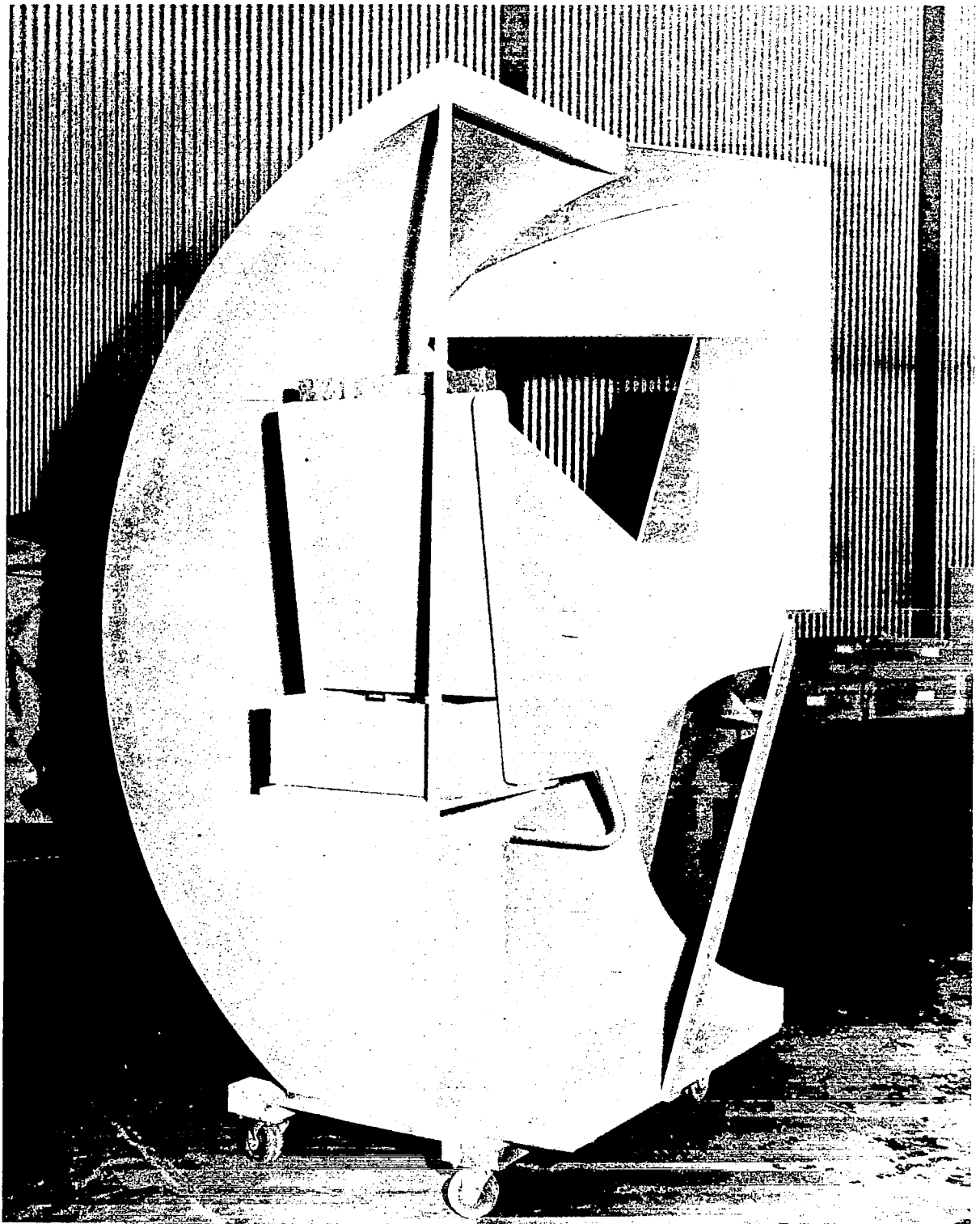
FIGURE B-2 LUNAR EXCURSION MODULE COCKPIT MOCK UP

# APPENDIX C

## DYNAMIC ANALYSES FOR LUNAR LANDING
## AND RENDEZVOUS

## C-1. LUNAR LANDING ANALYSIS

### A. INTRODUCTION

This section describes the use of simple images projected on the window to provide a flight path director and speed control display that is suitable for manual control of a lunar landing. The synthesis of such a display system for a vehicle that is dynamically similar to the LEM is discussed and the results of an analog simulation are presented.

Although the system described in this section is referred to as one for lunar landing, the same principles would apply to landing a vehicle on any planet or satellite that has significant gravity and negligible atmospheric density.

### B. DISPLAY DESCRIPTION

A manned lunar landing vehicle will almost certainly require a manual control mode for the terminal phase of its landing mission. Manual control would be the primary mode in exploratory missions and would provide a back-up capability for any automatically controlled system designed to land at prepared sites that are equipped with guidance beacons. Whether controlling the landing or monitoring an automatic system, the pilot's visual attention should be concentrated upon his view of the landing site rather than upon any instrumentation within the vehicle. In considering methods of increasing the information available to the pilot during the lunar landing, the application of projected display techniques that are being developed by Sperry for aircraft landing appears to be particularly appropriate.

The display configuration that has been studied is one designed for a constant path angle approach to a target on the surface, with programmed speed. Although this display study has been restricted to two dimensions, the concepts that have been developed can be applied to cross-course control as well. A director type of path display was postulated; that is, one that causes the vehicle to follow a desired glide path to the target if the pilot keeps a "path marker" superimposed on the target. Figure C-1 illustrates the display geometry and the two images that are projected on the window. The pilot's task is to keep the path marker on the target so as to minimize the angle $\Delta$ , and to keep the speed marker aligned with the path marker so as to minimize the angle $\sigma$ .

In order to study the synthesis of suitable driving functions for the two projected images, it was necessary to start with the assumption of a specific vehicle to permit analog computer simulation of system dynamics. A vehicle representative of the LEM was chosen with the assumption that its only translational thrust comes from a fixed, variable-thrust main propulsion engine, so that the vehicle has to be tilted to accelerate horizontally while landing. It was assumed that the vehicle was equipped with the following controls:

- Thrust Control - Continuously variable thrust proportional to throttle position, within thrust acceleration limits of 3 to 32 ft/sec$^2$.

- Pitch Control - Either pitch rate or pitch displacement commanded by stick displacement. In the pitch rate system a one second, first order lag between stick displacement and pitch rate was assumed. Maximum stick displacement commanded 10 degrees/sec. The displacement system assumed second order dynamics with a 0.5 damping ratio at a natural frequency of 1 radian/second and a maximum pitch command of 20 degrees.

The performance limits indicated in table C-1 were adapted rather arbitrarily as quantitative goals for landing. It is believed that they are typical of the values established for the LEM.

## TABLE C-1

### MAXIMUM ACCEPTABLE VALUES AT TOUCHDOWN

| | |
|---|---|
| Vertical velocity, $\dot{h}$ | -10 ft/sec |
| Horizontal velocity, $\dot{x}$ | ±5 ft/sec |
| Pitch attitude, $\Theta$ | ±2.5 degrees |
| Pitch rate, $\dot{\theta}$ | ±9 degrees/sec |

## C. SYSTEM SYNTHESIS

One of the ground rules adopted for this display study was that the system should employ a minimum number of sensors. This was established so that sensor redundancy would be practicable for reliability purposes if the display is a primary system, or to permit an independent set of sensors if the display is used as a back-up system.

The selection of a particular glide path angle and speed program to be used with the display is not obvious. Based upon aircraft and helicopter experience, pilot opinion appears to favor a relatively flat path angle, primarily for ease of judging terrain features. The flat approach would also be compatible with an efficient trajectory for descending from lunar orbit to the lunar surface, although a general proof would be well beyond the scope of this study. If the glide path angle is to be held constant until reaching the surface, then a steep angle would appear to be best suited to the relative magnitudes of the limiting touchdown velocities and the landing pad structure of the LEM type of vehicle. Another consideration is that in order to vary speed while descending at a constant path angle, the total acceleration vector must lie along the path. Using the notation of figure C-1 this can be expressed as

$$\frac{\ddot{h}}{\ddot{x}} = \tan G \qquad (C-1)$$

Since the main engine has been assumed to be the sole source of vehicle acceleration, the following will be true:

$$\tan \theta_r = \frac{\ddot{x}}{\ddot{h} + g} \qquad (C-2)$$

Combining these equations gives

$$\tan \theta_r = \frac{1}{\tan G + \dfrac{g}{\ddot{x}}} \qquad (C-3)$$

Thus the required pitch angle increases as the path angle decreases. This relationship is illustrated in figure C-2. In addition, this figure presents values of normalized thrust acceleration based upon the relation

$$\frac{a_T}{g} = \sqrt{\left(\frac{\ddot{x}}{g}\right)^2 + \left(\frac{\ddot{h}}{g} + 1\right)^2} \qquad (C-4)$$

The angle $(\theta_T + G)$, which defines the lower limit of the pilot's field of view required to keep the aiming point in sight (assuming zero engine trim, $\delta$), is also plotted in figure C-2, from the following:

$$\left[\frac{1}{2 \cos (\theta_r + G)}\right]^2 = \left[\frac{\ddot{x}}{g} - \frac{\tan (\theta_r + G)}{2}\right]^2 + \left[\frac{\ddot{h}}{g} + \frac{1}{2}\right]^2 \qquad (C-5)$$

It can be seen, therefore, that the limiting acceleration along a given path angle will depend upon the available thrust acceleration, the field of view, and the maximum allowable pitch attitude, any one of which may be the limiting factor.

In considering the speed program to be used in the descent, the theoretical optimum in terms of fuel consumption is a program that calls for the use of the largest possible thrust for the shortest possible time just before touchdown. The difficulty with this approach is the large transients in thrust and pitch that would have to be produced at very low altitude. This study assumed that it was important to minimize low altitude speed changes and consequently used an exponential speed-range law. That is, a reference rate of descent was computed as proportional to altitude plus a bias necessary to provide a finite rate of descent at touchdown. The computed altitude reference rate was then

$$\dot{h}_R = \dot{h}_T - \frac{h}{\tau_h} \tag{C-6}$$

where

$\dot{h}_T$ = desired altitude rate at touchdown

$\tau_h$ = arbitrary time constant in seconds.

If this reference rate is followed from some initial altitude rate, $\dot{h}_1$, corresponding to an initial altitude, $h_1$, the time to reach the surface is

$$t_T = \tau_h \ln\left(1 - \frac{h_1}{\tau_h \dot{h}_T}\right) = \tau_h \ln\left(\frac{\dot{h}_1}{\dot{h}_T}\right) \tag{C-7}$$

In addition, the necessary vertical acceleration of the vehicle can be expressed as

$$\ddot{h}_R = \frac{h}{\tau_h^2} - \frac{\dot{h}_T}{\tau_h} = -\frac{\dot{h}_R}{\tau_h} \tag{C-8}$$

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Examination of equations C-7 and C-8 indicates that while the selection of a short time constant will minimize the time to descend, it will also increase the necessary acceleration. During this study a time constant, $\tau_h$, of 50 seconds was adopted.

Two distinctly different display systems were synthesized during this study. They differ in that the one best suited for steep approach angles calls for controlling path angle through pitch and rate of descent with throttle, while the other, which is intended for flat approaches, reverses these control actions. Block diagrams of the two systems are shown in figures C-3 and C-4. These figures include the expressions for the vehicle dynamics and space geometry involved in the systems.

The system diagrams indicate that pitch attitude, altitude, altitude rate and vertical acceleration are quantities that must be sensed for display computation purposes, but the diagrams do not specify the sensors, which may be listed as

- Pitch Attitude - Vertical Gyro

- Altitude - Radar Altimeter

- Altitude Rate - Doppler radar or derived from altitude

- Vertical Acceleration - Platform mounted accelerometer; derived from altitude rate; or computed from body-mounted accelerometer and pitch information.

## D. SYSTEM PERFORMANCE

The steep and shallow approach angle systems are both based upon the same dynamic principles and share several features. By discussing the simpler steep angle system first, the shallow system will be more readily understood.

In the steep angle system the speed control display consists of an altitude rate error indication (the angle $\sigma$ between the speed and path marker images). The pilot's task is to minimize this angle by reducing throttle when speed is high, i.e., when the speed marker is above the path marker. With the 50 second altitude time constant and a display gain, $K_D$, on the order of 0.25 degree per foot/second, the system worked out very well. The path control is a director system in which the pilot's task is to keep the path marker image superimposed upon his chosen target by pitching up when the image is above the target. The angular relationship of the path marker image to the target is defined in figure C-1 as

$$\Delta \equiv E + \Theta - \lambda \qquad\qquad (C-9)$$

The path marker image position with respect to the vehicle is shown in figure C-3 to be

$$\lambda = \Theta_M + G + \ell \qquad\qquad (C\text{-}10)$$

By assuming that there is negligible error in the measurement of pitch, the two equations above can be combined to obtain

$$\Delta = E - G - \ell = \epsilon - \ell \qquad\qquad (C\text{-}11)$$

Thus when the lead angle, $\ell$ is equal to the path error, $\epsilon$, the path marker will be on the target. By choosing a large value for the lead angle time constant, $\tau_G$, the lead angle becomes dynamically similar to pitch integral, which in turn is similar to horizontal velocity and rate of change of elevation angle. By keeping $\Delta = 0$, path error will be wiped out at a rate approximately proportional to the error. By making the lead angle gain a function of the selected path angle, G, and also making it an inverse function of altitude as shown in figure C-3, the path error behavior for small errors can be shown to be characterized by second order dynamics with the following approximate values:

$$\text{Natural frequency,} \quad \omega_G \approx \sqrt{\frac{1}{K_G}} \qquad\qquad \text{radian/second} \qquad (C\text{-}12)$$

$$\text{Damping ratio,} \quad \zeta_G \approx \frac{\tau_G\, \omega_G}{2} \qquad\qquad (C\text{-}13)$$

It was found in the simulation program that the displacement type of pitch control system made this path control much easier to track than the pitch rate control system did. (With the pitch rate system the inner loop tends to be unstable.) It was also found desirable to include the vertical acceleration input to the lead angle computation that is shown in figure C-3. This input commands the pitch angle approximately necessary to maintain the desired path angle without actually incurring a path error; it implicitly satisfies equation (C-2).

The display simulation was accomplished by analog computation of the image angles, $\Delta$ and $\sigma$, by using them to drive a pair of pointers vertically. The pilot flew the display by commanding pitch and throttle so as to keep the pointers aligned with a fixed point that represented the target.

Figure C-5 is a typical time history of a simulated descent with a 63.4 degree glide path (tangent G = 2.0). The run started with the following initial conditions:

$$h_1 = 5000 \text{ feet} \qquad\qquad X_1 = 2500 \text{ feet}$$

$$\dot{h}_1 = -103 \text{ feet/sec} \qquad\qquad \dot{X}_1 = -51.5 \text{ feet/sec}$$

$$a_{T_1} = g = 5.31 \text{ feet/sec}^2 \qquad\qquad \Theta_{T_1} = 0 \text{ degree}$$

The path angle dynamics were set for a natural frequency of 0.025 radians/ sec and damping ratio of 1.0; pitch displacement control was being used.

While this run shows well-behaved pitch attitude, it can be seen from figure C-3 that if the glide path angle, G, is reduced, the sensitivity of lead angle to pitch activity consequently increases. This condition led to the synthesis of the shallow angle system of figure C-4, in which the pilot uses throttle control to stay on the path and pitch control to vary speed. The description of the director function of the path control that was given above for the steep angle system applies here also, except that the pilot reduces thrust when the image is above the target. Thus a vertical velocity approximately equal to path error is produced and the error is wiped out with a proportional rate.

The speed control of the shallow angle system introduces a director function for the speed marker image. If the pilot pitches down when the speed marker image is above the path marker, and so keeps the angle at zero, he thereby makes a pitch angle, which is dynamically similar to horizontal acceleration and is approximately equal to vertical rate error. If it is assumed that the proper path angle is followed closely, this means that vertical acceleration will be proportional to horizontal acceleration and consequently to vertical rate error. It was found desirable to include the pitch attitude input in the path marker lead angle computation to satisfy equation (C-2). This input serves to prevent speed control action from disturbing the path and results in a lead angle computation that is dynamically of exactly the same form as that used for the steep angle system, although it appeared desirable to use somewhat different parameters.

The natural frequency and damping ratio expressions for the path angle dynamics given in equations (C-12) and (C-13) apply to the shallow angle system as well. During the simulation program, values of 0.05 radians/sec

for natural frequency and 0.5 for damping ratio were found to work well. The dynamics of the speed control loop are somewhat more complex, having a third order characteristic equation. The open loop transfer function for the speed loop can be represented as follows for small errors:

$$\left(\frac{1}{K_s\, s}\right) \quad \left(\frac{r_h\, s + 1}{r_h\, s}\right) \quad \left(\frac{r_s\, s + 1}{r_s\, s}\right)$$

In the simulation work, values of 20 seconds for $r_s$ and 5 for $K_s$ were found compatible with the 50-second $r_h$ that had been previously chosen. These values give a closed loop equation characterized by a 55 second lag and a natural frequency of 0.105 radians/sec with 0.8 damping. Root-locus analysis indicates that the stability of this loop is gain-dependent, with the criterion for a stable loop being

$$K_s < r_s + r_h = 70 \tag{C-15}$$

Therefore, the gain used in the simulation had a comfortable stability margin. The actual root-locus plot is shown in figure C-6, with several values of $K_s$ identified.

A typical simulation run for the shallow angle system on a 26.6-degree glide path (tangent G = 1/2) is given in figure C-7. This run started with the following initial conditions:

$h_1 = 2500$ feet $\qquad\qquad x_1 = 5000$ feet

$\dot{h}_1 = -52.5$ feet/sec $\qquad\qquad \dot{x}_1 = -105$ feet/sec

$a_{T_1} = 6.3$ feet/sec$^2$ $\qquad\qquad \Theta_{T_1} = 0$ degree

It can be seen from figure C-7 that both pitch and thrust activity are small at low altitudes. The vertical and horizontal rates at touchdown were very close to the nominal values of -2.5 and -5.0 feet/sec respectively. This run used the pitch rate control system, which appeared to be capable of results that equaled the displacement system. The chief difficulty encountered

with the rate system during the simulation program was that while in the initial transient stage of the run, it was very easy to produce large pitch angle. Since the display computations are based upon the assumption of pitch angles that do not, in general, exceed 25 or 30 degrees, it is necessary to monitor pitch attitude or use some pitch limit device while employing the pitch rate control. In an actual display implementation, attitude information would be furnished and would significantly increase the ability to fly the approaches over that attainable in the simple simulation pointer matching display. Inclusion of attitude cues would also affect the relative effectiveness of pitch displacement and pitch rate control schemes.

## E. STEEP VS SHALLOW APPROACH SYSTEM COMPARISON

The advantages of each of the two approach systems can be summarized as follows:

Shallow approach system:

- Low altitude pitch maneuver requirements are small

- Pitch rate control mode is satisfactory

- Tighter path dynamics appear to be possible

Steep approach system:

- Horizontal speed can be more closely controlled by selecting a steep path

- Large pitch angles are not required. Use of pitch displacement control readily permits pitch limiting

- Path and speed control can each be exercised independently, if desired.

These conclusions are based on the rather limited simulation program and are considered conservative. Further simulation with a more realistic display which includes at least attitude information is necessary for final system definition and conclusions.

## F. INITIAL CONDITION CONSIDERATIONS

Both steep and shallow angle systems were studied in the simulation program with initial path and velocity errors. It was apparent that there were limiting values of initial errors beyond which system performance would not be satisfactory unless the speed program was modified in some fashion to permit sufficient time to correct path errors before reaching the surface.

One question that arises in considering the initial condition problem is whether it is necessary to preselect the glide path angle, G, or can it be so adjusted by the pilot at the initiation of the descent program as to reduce initial path errors? Note that a large range of path angle selection would probably necessitate variable system gains. An alternative consideration is the possibility that, instead of a predetermined target, the pilot might be free to select an aiming point that is compatible with an arbitrary glide path.

It has been tacitly assumed here that the manual landing control (approach) is preceded by a period of automatic path control (initial ascent), using inertial, and possibly radar, guidance techniques. The projection display capability might well be used in the automatic mode to indicate the aiming point to the pilot and so permit him to monitor the guidance system and perhaps enable him to superimpose inputs to change the aiming point if he desires. This would reduce the initial condition problem that would have to be considered in the manual system design.

## G. SUGGESTED EXTENSIONS OF STUDY

The basic study work that has been conducted to date has served to show that a dynamically acceptable display system can be synthesized for the lunar landing approach. A number of possible extensions of this basic work can be suggested as follows:

- Develop a rudimentary visual display for a simulation program that would include an indication of the horizon and target size and shape. This would allow visual softening of the pilot's actions as range decreases and would permit much more meaningful pilot opinion than can be obtained from the pointer-matching display that has been used to date.

- Simulate the use of alternate sensors in suitably modified systems. Thus three-axis doppler radar might be considered in order to replace pitch with horizontal velocity in the path marker lead angle computation. The effects of typical noise in radar and accelerometer outputs could also be simulated.

- Expansion of the simulation program to include lateral path control and a two-degree-of-freedom path display.

- The application of the manual control system to the terminal portion of a specific initial descent guidance scheme. This would permit a realistic study of initial conditions for the manual control phase and possible monitor and control super-position methods for combined manual/automatic control.

# C-2. RENDEZVOUS GUIDANCE AND CONTROL STUDIES

## A. TECHNIQUES

Rendezvous guidance or steering techniques can be catalogued into two general categories: those techniques based on proportional or constant bearing navigation and those based on orbital mechanics. In the proportional navigation technique, the angular rate of rotation of the velocity vector with respect to inertial space is controlled in proportion to the angular rate of the line of sight and gives an exponential approach to a constant bearing course. This technique is generally used in the terminal phase of rendezvous where the thrust is continuously programmed to bring the range and range rate to zero simultaneously.

In the orbital mechanics techniques, solutions to the equations of motion between the rendezvous (RV) vehicle and its target are used to compute a trajectory (collision orbit) which will lead to an interception between the two bodies at some future time. This technique consists of an acceleration phase, which produces the required velocity change, a coasting phase, and a deceleration phase starting at some time prior to rendezvous. The orbital mechanics technique is more applicable for prediction where large changes occur during the interception phase. This technique is more optimum in terms of fuel consumption and rendezvous time. However, the proportional navigation technique is more suitable during the final stages of rendezvous where precise matching of vehicle kinematics and attitudes are required. In this case, considerations of minimum fuel, minimum time, etc., can be subordinated to fine vernier control.

## B. PROPORTIONAL NAVIGATION CONTROL LAWS

A guidance law for velocity and range control based upon a modification of the proportional navigation theory is proposed by Cicolani (reference 5). The vehicle velocity is reduced to zero as the vehicle passes through the desired intercept point. The vehicle acceleration, $a_c$, which is commanded to follow a prescribed function based upon a feedback of the measured vehicle motion is given by the relation

$$a_c = \frac{V^2}{R} \left[ (S - 2) \cos L - \left( S - \frac{K+1}{K} \right) \frac{\sin L}{L} \right] \underline{1}_v + S \left( \omega_{LS} \underline{\times V} \right)$$

$$(C-16)$$

where $\underline{V}$ is the vehicle velocity, R is the range to the target, $\omega_{LS}$ is the angular velocity of the line-of-sight, L is the lead angle of the velocity vector

relative to the line-of-sight, and $\underline{1v}$ is a unit vector along $\underline{V}$. The quantities S and K are guidance system parameters whose numerical values must both be greater than 2 in order for the velocity to be reduced at rendezvous without requiring an unreasonably large final thrust.

The first term of the preceding equation controls the deceleration of the vehicle to reduce the velocity to zero at rendezvous. The second term is the conventional proportional-navigation relation used to place the vehicle on a trajectory which intercepts the target. For reasonable small values of initial lead angle, the first factor remains essentially constant throughout the flight, having a numerical value approximately equal to $\left( \dfrac{1 - K}{K} \right)$. This guidance law does not restrict the terminal-phase maneuver to a reference trajectory, but provides complete flexibility of control for all initial conditions within the capabilities of the vehicle propulsion system.

Large values of S increase the rate at which the lead angle is decreased to zero, and, therefore, require a greater acceleration at the start of the flight. This will result in an increase in the initial thrust. The time of flight, $t_f$, for the terminal phase is directly proportional to K:

$$t_f = \frac{K R_o L_o}{V_o \sin L_o} \qquad (C\text{-}17)$$

The zero subscript refers to initial conditions. Since the component of required thrust normal to the vehicle velocity vector is reduced to zero early in the flight, the time of flight in the preceding equation is relatively insensitive to the initial lead angle, $L_o$.

Typical rendezvous maneuvers are illustrated in figures C-8, C-9 and C-10. Figure C-8 is a dimensionless polar plot of paths in the plane of relative motion $\left( \dfrac{R}{R_o} \text{ vs } \gamma \right)$. The origin is the origin of the reference frame, and the vertical radial line is the initial line-of-sight direction. The equation governing these paths is

$$\frac{R}{R_o} = \left[ \frac{\sin \left[ L_o - (S - 1)\gamma \right]}{\sin L_o} \right]^{\frac{1}{S-1}} \qquad (C\text{-}18)$$

The paths, therefore, depend only on $L_o$ and S. Since the maneuver time is independent of S, a set of values of S will give a set of paths of constant maneuver

time for given values of $L_o$, $R_o$, $V_o$, and K. Changes in K do not affect the path for a given value of S but change only the time required to traverse the path.

Polar plots of $\frac{V}{V_o}$ versus L for various values of S and the initial condition $L_o = \frac{\pi}{2}$ are shown in figure C-9. The equation is

$$\frac{V}{V_o} = \left(\frac{\sin L}{\sin L_o}\right)^{\left(\frac{S-2}{S-1}\right)} \left(\frac{L}{L_o}\right)^{\left[S - \frac{K+1}{K}\Big/S-1\right]} \tag{C-19}$$

Plots of lead angle versus velocity for several values of K are given in figure C-10.

The vehicle engine for the above maneuvers is required to provide a command thrust proportional to the sum of the command acceleration plus the tidal acceleration (differential gravitational acceleration between the two vehicles). However, Hord (reference 6) has demonstrated that for satellites at altitudes from 100 to 4000 miles with terminal ranges of less than 100 miles, and rocket thrusting of 5 to 10 g's, the effects of the gravitational field upon both vehicles is nearly equal and therefore these differential gravity effects can be neglected. This approximation improves as the relative range between the vehicles approaches zero.

## C. LOGARITHMIC GUIDANCE

Under the condition that the initial lead angle, $L_o$, is zero; that is, the vehicle is thrusting along the line-of-sight, equation (C-16) reduces to

$$a_c = \frac{V^2}{R}\left[\frac{1-K}{K}\right] = k\frac{\dot{R}^2}{R} = \ddot{R} \tag{C-20}$$

This control function, when integrated, provides the following relationship among range, range rate, the constant k, and the initial conditions:

$$\frac{d(\dot{R})}{\dot{R}} = \frac{kdR}{R}$$

$$\log\left(\frac{\dot{R}}{\dot{R}_o}\right) = k\log\left(\frac{R}{R_o}\right)$$

$$\frac{\dot{R}}{\dot{R}_o} = \left(\frac{R}{R_o}\right)^k \tag{C-21}$$

The logarithmic character of (C-21) as the primary solution of (C-20) is interesting since this type of relationship is one of the simplest which provides acceptable terminal behavior completely independent of k and the initial conditions. This simplified form of equation (C-16) has been further analyzed by Green (reference 7) and has been termed "Logarithmic" guidance. It is also interesting to note that the lead angle in control equation (C-16) is decreased to zero early in the flight and equation (C-20) is applicable for the final portion of the rendezvous maneuver.

## D. ERROR CONSIDERATIONS FOR LOGARITHMIC GUIDANCE

Green (reference 7) has shown that a logarithmic guidance system can cope with errors which change the desired accelerations by nearly a factor of two and still attain the desired terminal conditions. The type of errors considered are scale factor errors, noise, limiting errors, and most other types of scaler errors. The effects of bias errors on the system were also investigated and the following effects were noticed.

- Bias decelerations which add to the desired deceleration cause the final rate to go to zero before the range does, while those bias decelerations that subtract allow zero range rate at impact. If the velocity goes to zero before impact, the guidance law will cause undesirable behavior during the remaining portion of the flight.

- Bias errors in the determination of the range rate which add to the magnitude of the actual range rate cause zero rate to occur at some non-zero range (assuming no negative range rates are permitted); bias errors which subtract cause impact at non-zero rates.

- Subtracting bias errors in range determination produces zero velocity at non-zero ranges, while additive errors provide non-zero impact range rates.

The effects of dynamic response errors upon logarithmic guidance were demonstrated through analog computer studies. The conclusions from these investigations are:

- Dynamic time lags between desired acceleration and actual acceleration of less than 2 percent of total time of flight will not affect the terminal behavior of the system.

- Time lags of nearly 5 percent of ideal time of flight can be tolerated between the measured range rate and the actual range rate without interference with the desired terminal conditions. However, these dynamic errors in range rate measurements cause a considerable increase in the actual flight time of the system. This increased flight time would reduce the systems' efficiency and force a compromise between the economy and the narrow bandwidths for sensor instrumentation.

- Dynamic errors in range determination are much more critical than the other dynamic errors studied. Any dynamic error in range produces a deviation from the ideal terminal conditions and a finite range rate exists at impact. However, for time lags of less than 2 percent of the time of flight, the impact range rates are less than 0.1 percent of the initial range rate.

## E. ORBITAL MECHANICS RENDEZVOUS TECHNIQUE

In the orbital mechanics technique, also known as the "two impulse" and "velocity impulse" technique, the transfer maneuver is performed by the application of two thrust impulses. One impulse places the RV vehicle on a collision trajectory; the other impulse, near the completion of the maneuver, reduces the contact velocity to zero. The trajectory between impulses is unpowered and can be computed from the equations of motion. The direction and magnitude of the velocity impulse to initiate the maneuver is a function of the initial position and velocity of the RV vehicle and target in their respective orbits and the desired rendezvous point. The criterion for selecting a desired rendezvous point may be based on considerations of least fuel, least energy, time, or preselected spacial positions. The application of the terminal phase is also based on the above considerations. Also to be considered are the acceleration capabilities of the propulsion system and the instrumentation errors in the guidance system. From a viewpoint of efficient fuel utilization, it is desirable to apply the major deceleration pulse at the latest possible time. This tends to keep the duration of the terminal phase small and, thus, minimizes the effect due to the differential gravitational field between the two vehicles. The later the terminal pulse is applied, however, the larger is the required engine thrust and vehicle acceleration. The optimum terminal phase initiation time thus involves a compromise between fuel considerations, the effects of large vehicle accelerations, and probability of collision.

Figure C-11 shows the geometry for a typical two-impulse rendezvous maneuver. The relationship between the geometric range, $R_2$, and range angle, $\Theta$, at the ignition point, C, and the burnout parameters $R_1$, $V_1$, and $\phi$ at the burnout point B is given by

$$\frac{R_1}{R_2(\theta)} = \frac{GM}{R_1 V_1^2}\left(\frac{1 - \cos\theta}{\cos^2\phi}\right) + \frac{\cos(\phi + \theta)}{\cos\phi} \tag{C-22}$$

The time of flight, $t_b$, is given by

$$t_b = \frac{R_1}{V_1 \cos\phi} \int_0^{\theta(t)} \left[\frac{GM}{R_1 V_1^2}\left(\frac{1 - \cos\sigma}{\cos^2\phi}\right) + \frac{\cos(\phi + \sigma)}{\cos\phi}\right]^{-2} d\sigma \tag{C-23}$$

These equations are derived from the differential equations of orbital motion. GM is the earths gravitational constant. For $\dfrac{R_1 V_1^2}{GM} < 1$, it can be shown that equation (C-22) is the polar equation of an ellipse about the earths center written in terms of the burnout parameters rather than the astronomical parameters. From equation (C-23) it is noticed that the $t_b$ is a dependent variable, and if pre-specified, must be solved for implicity.

Assuming a constant thrust engine is to be used for the terminal phase pulse, then one way of applying the terminal pulse, which is relatively efficient from the fuel utilization viewpoint, is to thrust in a fixed direction (with respect to inertially-fixed coordinate frame) throughout the pulse interval. The required pulse velocity increment, $\Delta V_T$, is related to the horizontal and vertical components of present vehicle velocity ($V_H$ and $V_V$) by the expression

$$\Delta V_t = \sqrt{\left(V_V + gt_c\right)^2 + V_H^2} \tag{C-24}$$

where g is the gravitational acceleration and $t_c$ is the engine burning time that would be required to obtain $\Delta V_T$. The burning time, $t_c$, is related to the engine thrust-to-weight ratio at the start of the terminal pulse $\left(\dfrac{F}{W_o}\right)$ by the expression

$$t_c = \frac{I_{sp}}{F/W_o}\left[1 - \exp\frac{-\Delta V_T}{gI_{sp}}\right] \tag{C-25}$$

where F and $W_o$ are in pounds, and $I_{sp}$ is the specific impulse in seconds. Fuel consumption is given by the expression

$$\Delta W = \frac{F\, t_c}{I_{sp}} \qquad\qquad (C\text{-}26)$$

The horizontal and vertical components of distance the rendezvous vehicle would travel during the pulse interval are given the the relations

$$R_H = V_H t_c - g\, I_{sp} t_c \, \cos \gamma \left[ 1 - \frac{\left( \Delta V_T / g\, I_{sp} \right)}{\exp \left( \dfrac{\Delta V_T}{g\, I_{sp}} \right) - 1} \right] \qquad (C\text{-}27)$$

$$R_V = V_v t_c + \frac{g t_c^2}{2} - g\, I_{sp} t_c \, \sin \gamma \left[ 1 - \frac{\left( \Delta V_T / g\, I_{sp} \right)}{\exp \left( \dfrac{\Delta V_T}{g\, I_{sp}} \right) - 1} \right] \qquad (C\text{-}28)$$

The orientation of the thrust vector with respect to the local horizontal, $\gamma$, is given by the relation

$$\gamma = \tan^{-1} \left( \frac{V_v + g t_c}{V_H} \right) \qquad\qquad (C\text{-}29)$$

The procedure for determining the time of initiation of the terminal phase involves the following steps: first, measurements are made of the components of present vehicle velocity $V_v$ and $V_H$. Using appropriate values of $I_{sp}$ and $\frac{F}{W_o}$ for the terminal stage propulsion system and an average value of g, equation (C-24) and (C-25) are solved simultaneously to give the computed required burning time $t_c$ and velocity increment $\Delta V_T$. These values are used in equations (C-27) and (C-28) to compute the distance that the vehicle would travel during the pulse interval ($R_c$), which is equal to $\sqrt{R_H^2 + R_V^2}$ .

When the difference between the measured distance from the vehicle to the selected thrust termination point ($R_m$) and the computed distance ($R_c$) is a minimum, the engine is ignited for the terminal phase pulse. Thrust termination takes place when the present measured velocity $V_t$, which is

equal to $\sqrt{V_H^2 + V_v^2}$, reaches a minimum value.

The terminal phase guidance system described above is essentially an open-loop system, in contrast to the previously described proportional navigation system, which is a closed loop system. As a result, it is important that the initial conditions for the second pulse be accurately determined, in order to avoid excessively large velocities at time of rendezvous. It is also important that uncertainties in the vehicles propulsion system, i.e., thrust and mass flow rate, be small since no specific thrust magnitude control is provided. These propulsion system uncertainties can be corrected if a limited throttling range is provided for the terminal phase engine. To compensate for the effects of sensor propulsion system uncertainties, and to assure a successful rendezvous, the nominal terminal stage burnout point can be biased by certain distance, i.e., 500 to 1000 feet. The remaining part of the mission can then be accomplished by using logarithmic guidance.

$$\Delta \equiv E + \theta - \lambda$$
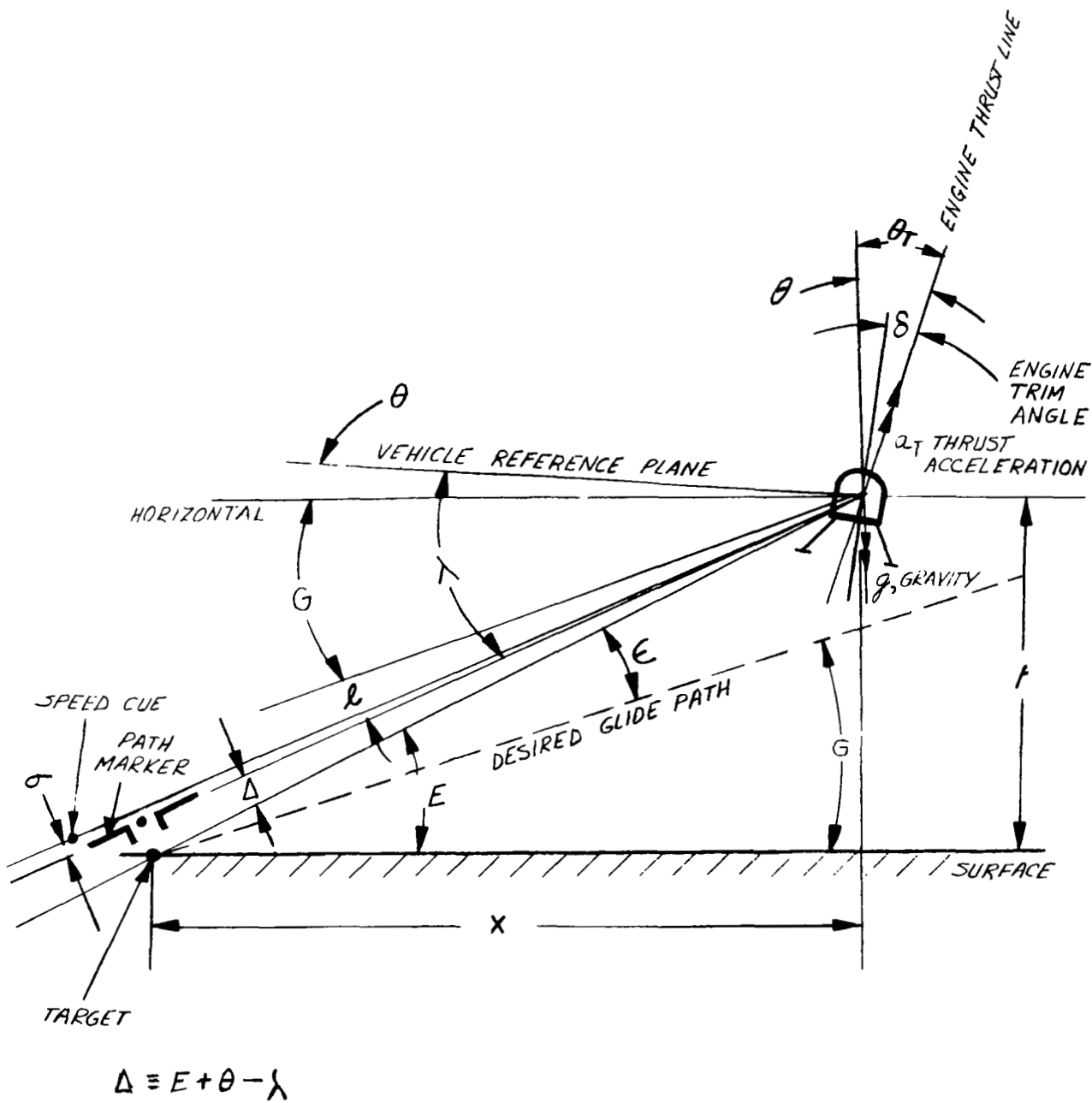
FIGURE C-1. LUNAR LANDING DISPLAY AND SPACE GEOMETRY
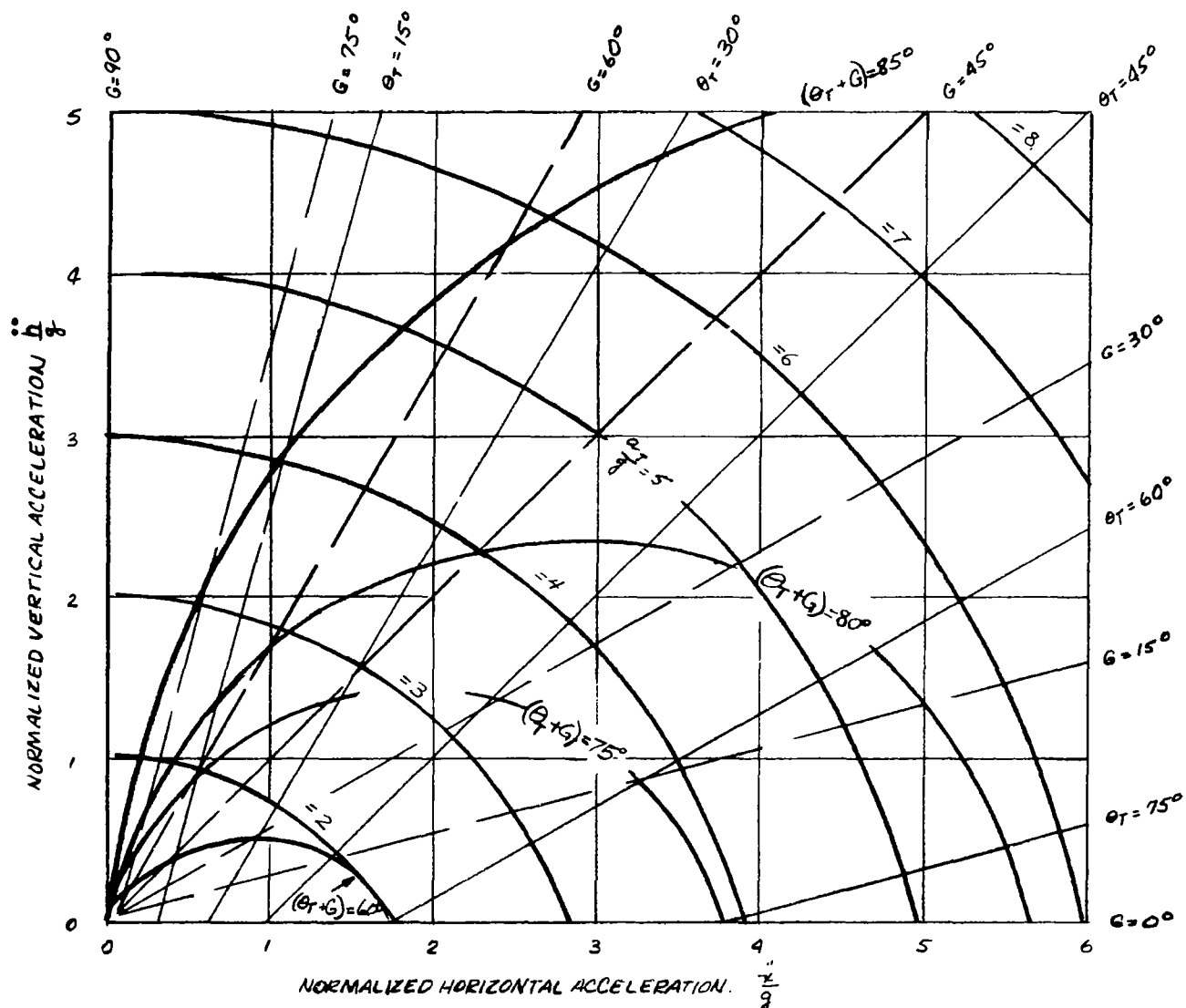
FIGURE C-2. PITCH ATTITUDE, THRUST ACCELERATION AND
FIELD OF VIEW REQUIREMENTS FOR ACCELERATIONS
ALONG CONSTANT GLIDE PATH ANGLES
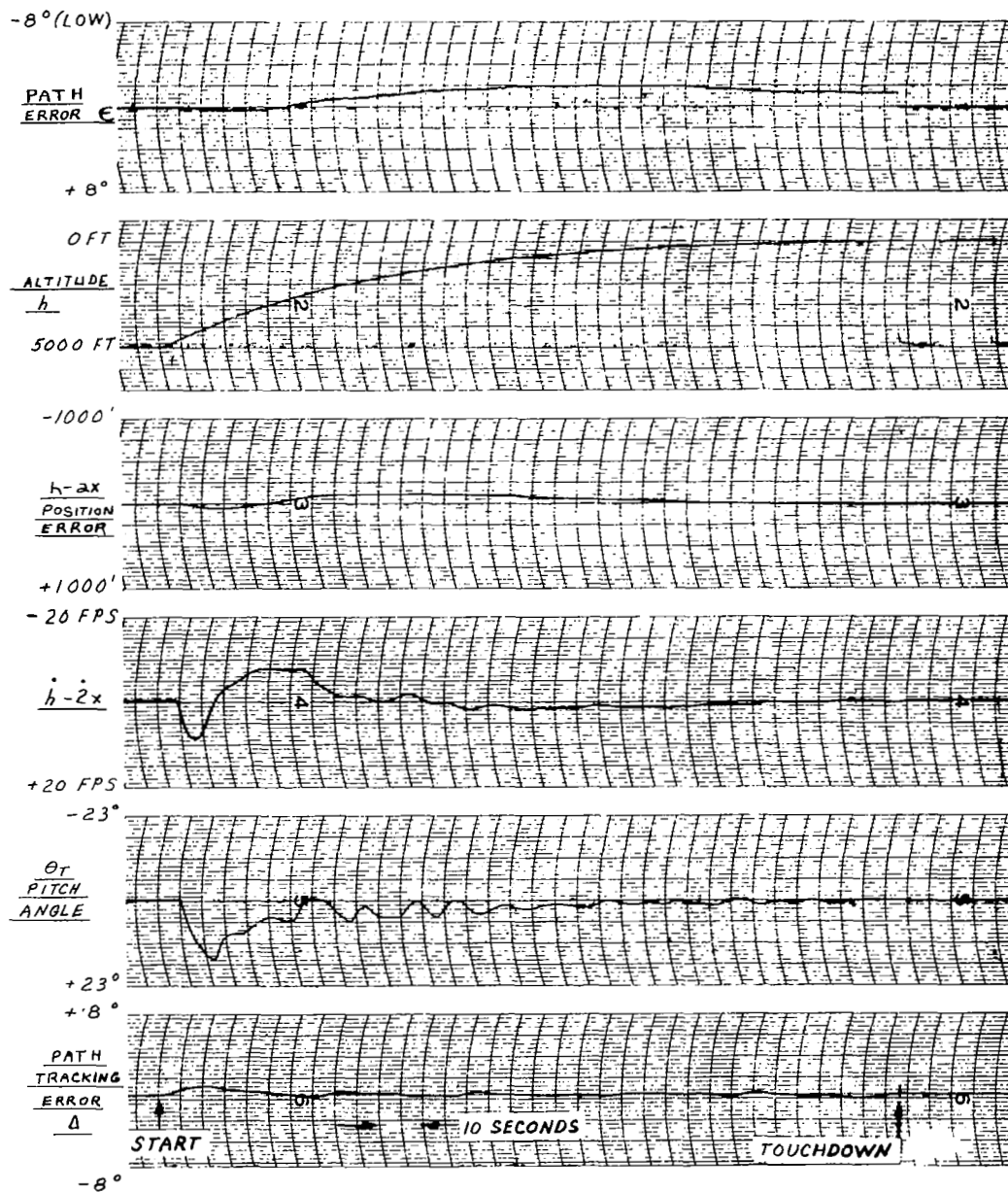
FIGURE C-3. STEEP PATH ANGLE APPROACH SYSTEM
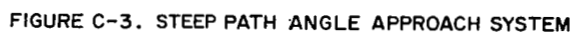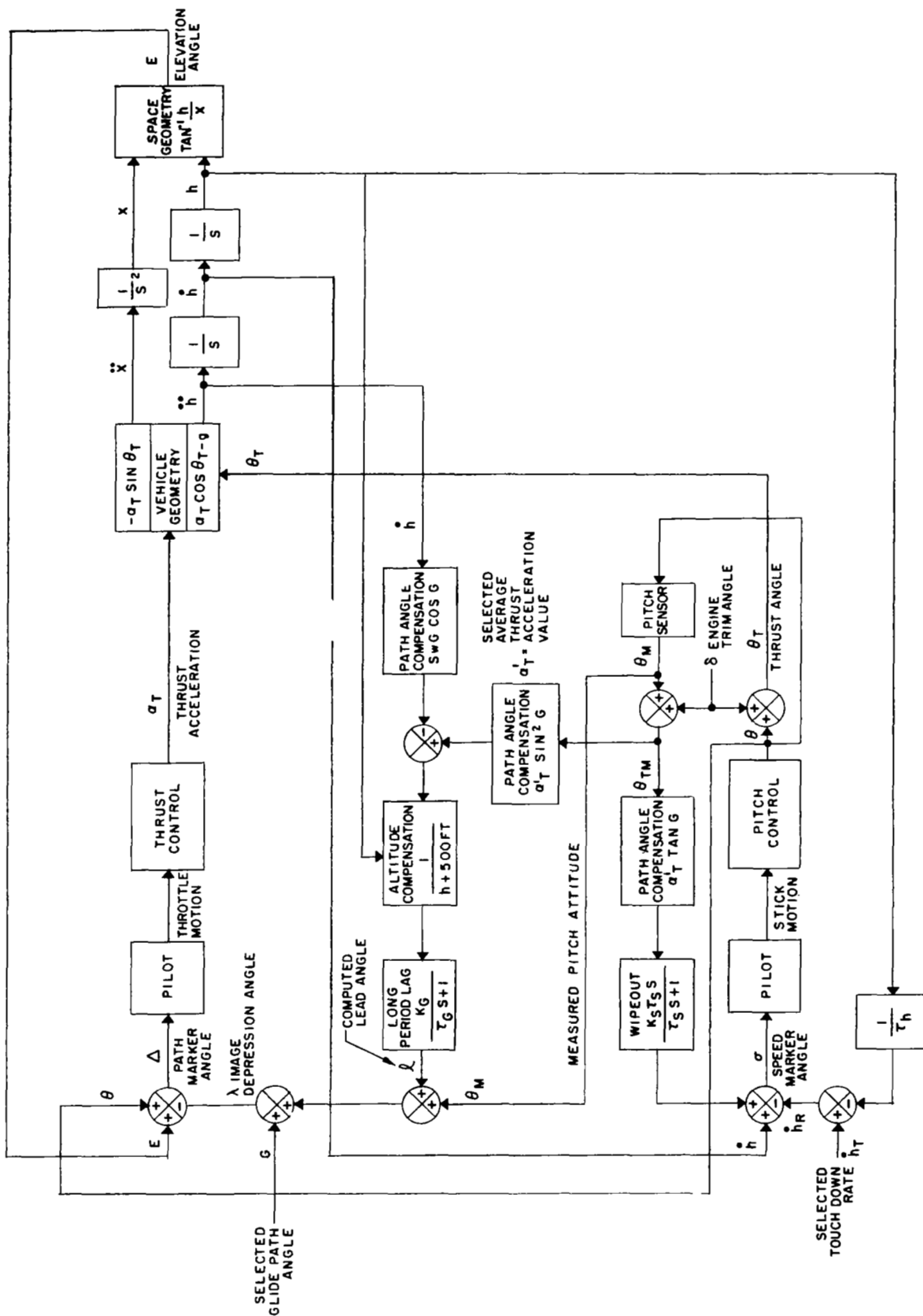
FIGURE C-4. SHALLOW PATH ANGLE APPROACH SYSTEM

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FIGURE C-5. TIME HISTORY OF STEEP (63.4°) ANGLE DESCENT

OPEN LOOP TRANSFER FUNCTION = $\dfrac{1}{K_s S}\left(\dfrac{20\,S+1}{20\,S}\right)\left(\dfrac{50\,S+1}{50\,S}\right)$

NOMINAL OPERATING GAIN
$K_S = 5.0$

$K_S = 11.4$

$K_S = 17.5$

$K_S = 7.0$

$K_S = 4.01$

TRIPLE POLE
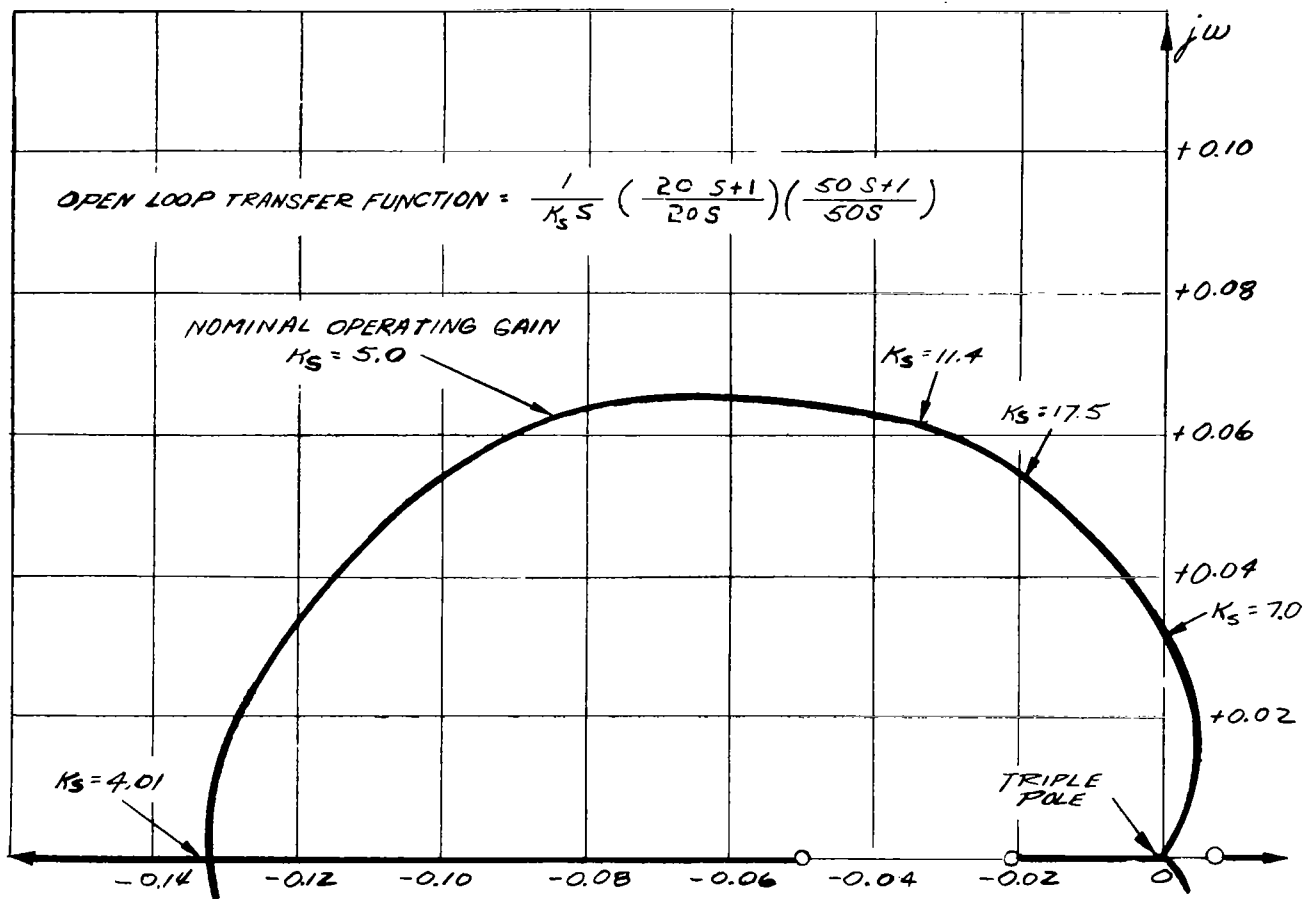
FIGURE C-6.  ROOT LOCUS PLOT OF SPEED CONTROL LOOP
FOR SHALLOW APPROACH SYSTEM
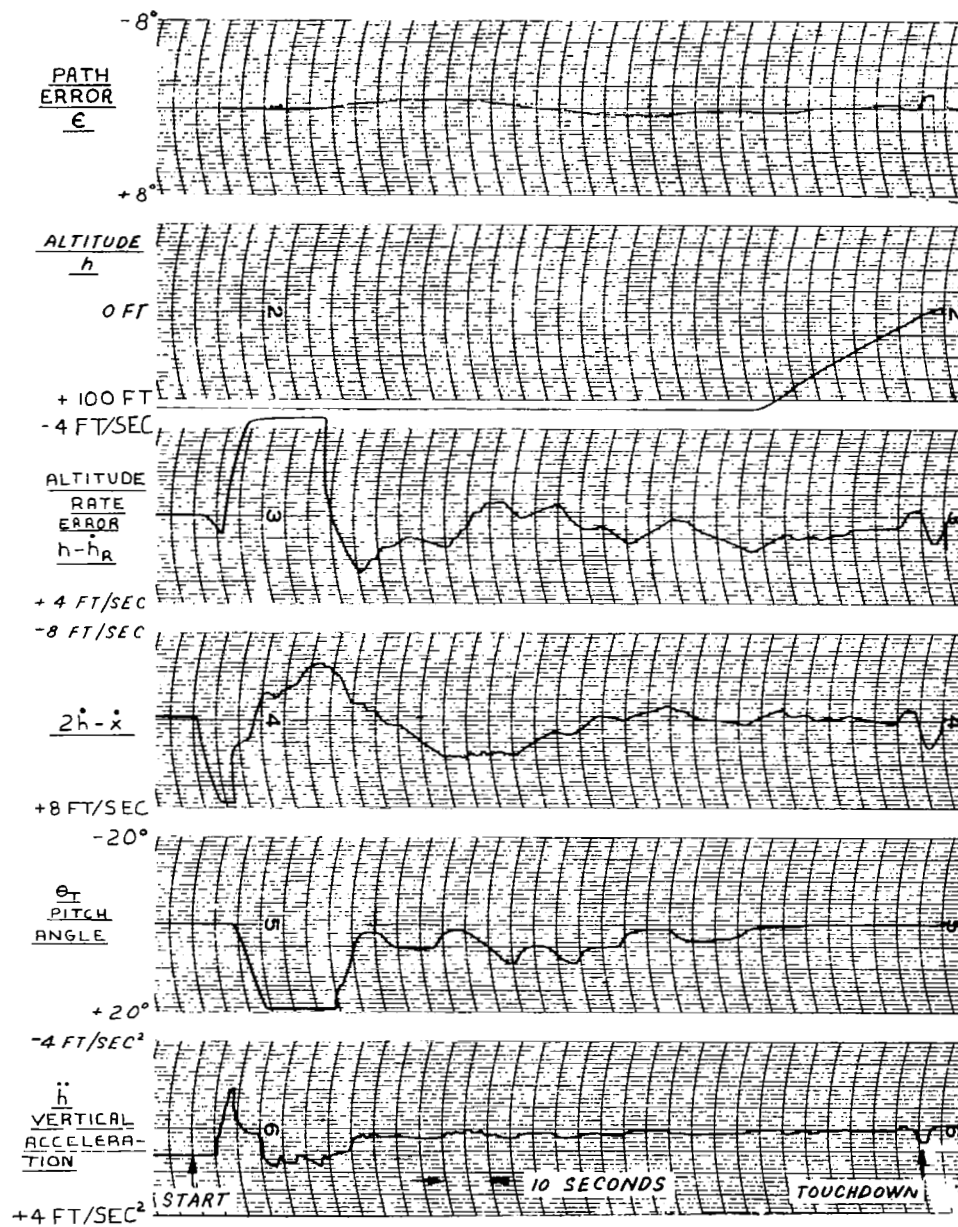
FIGURE C-7. TIME HISTORY OF SHALLOW (26.6°) ANGLE DESCENT
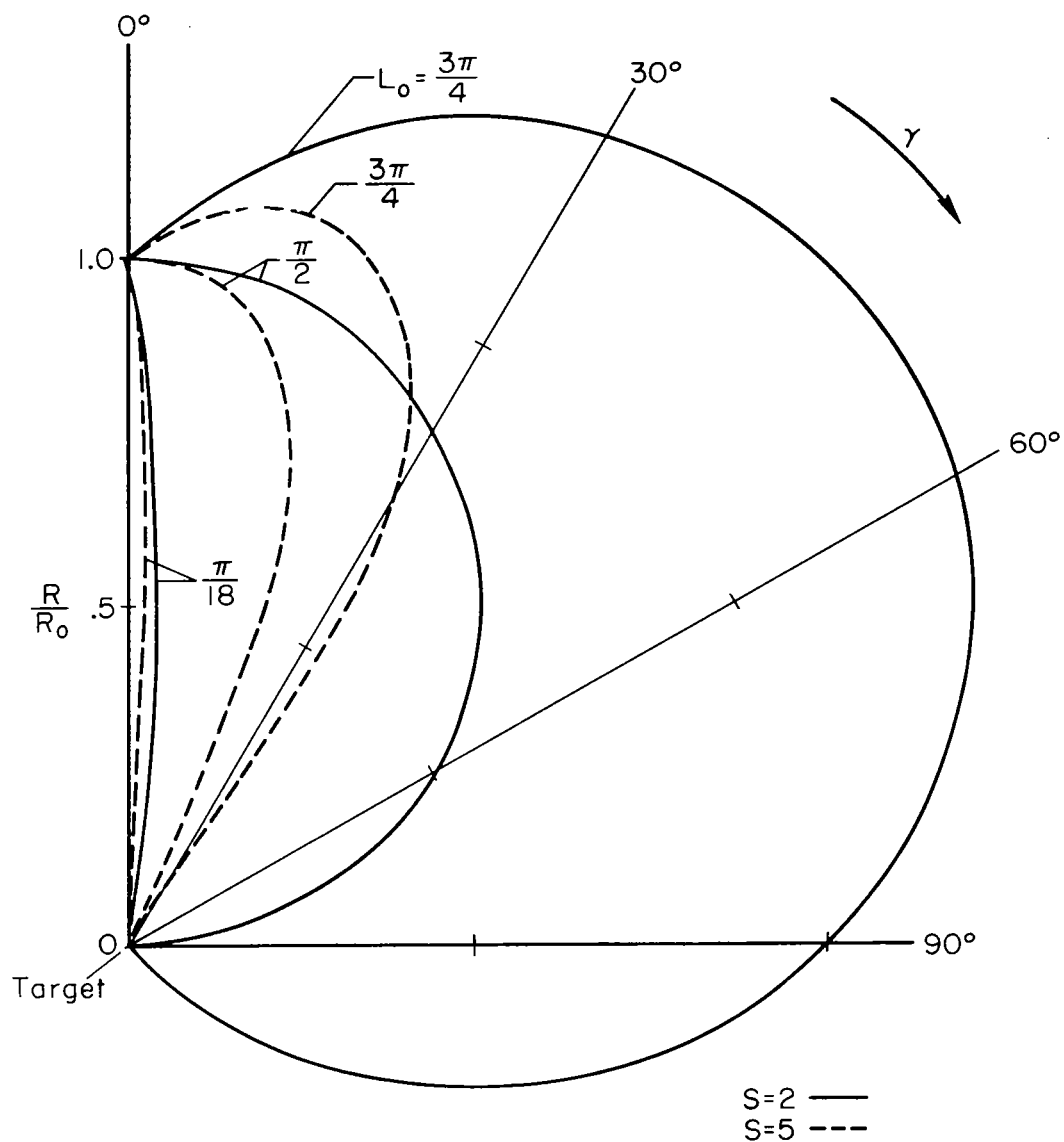
FIGURE C – 8.    MANEUVER PATHS: RANGE VERSUS ANGULAR DEPARTURE
FROM THE INITIAL LINE OF SIGHT
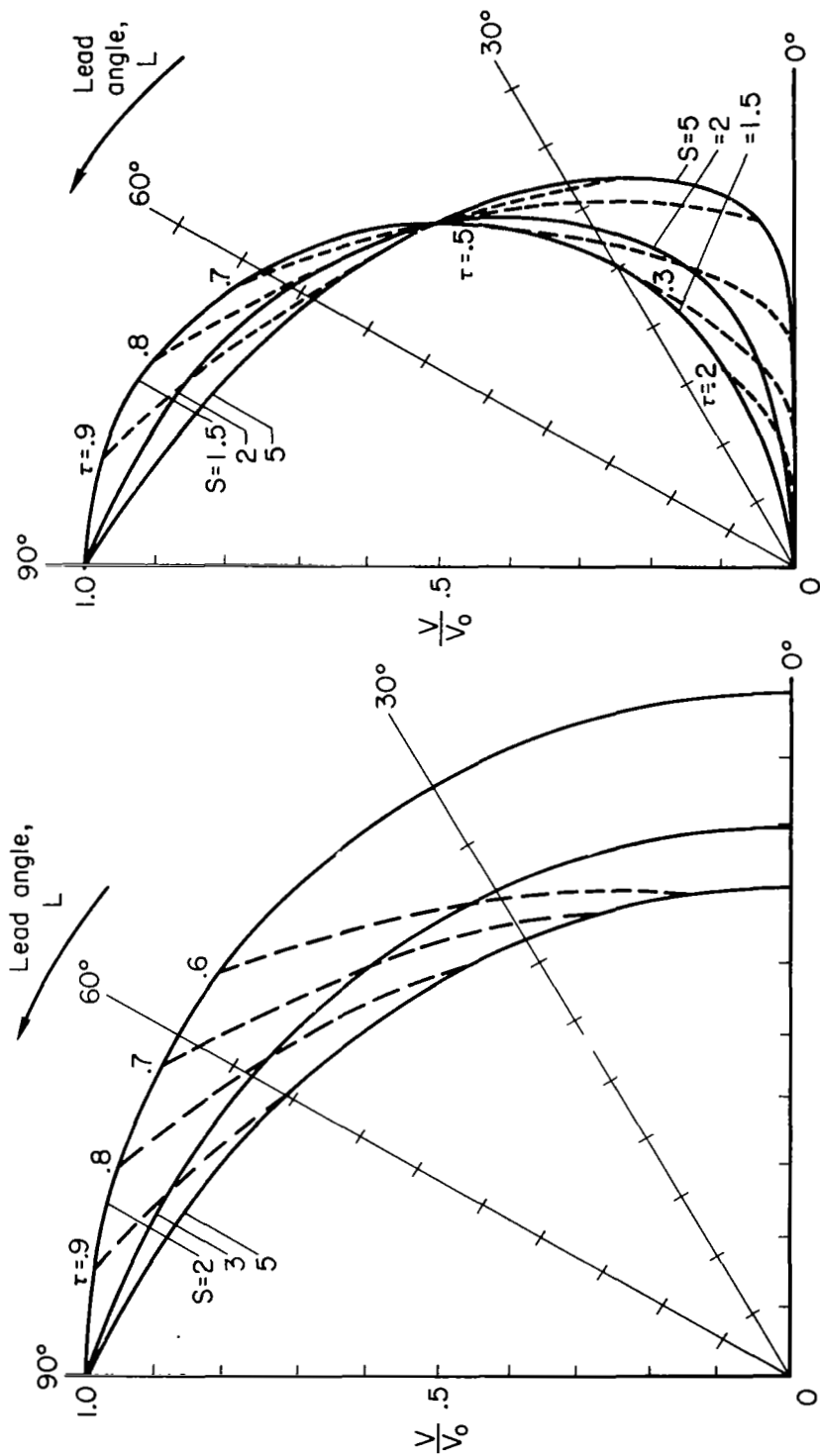
FIGURE C — 9. LEAD ANGLE VERSUS VELOCITY, $L_0 = \frac{\pi}{2}$
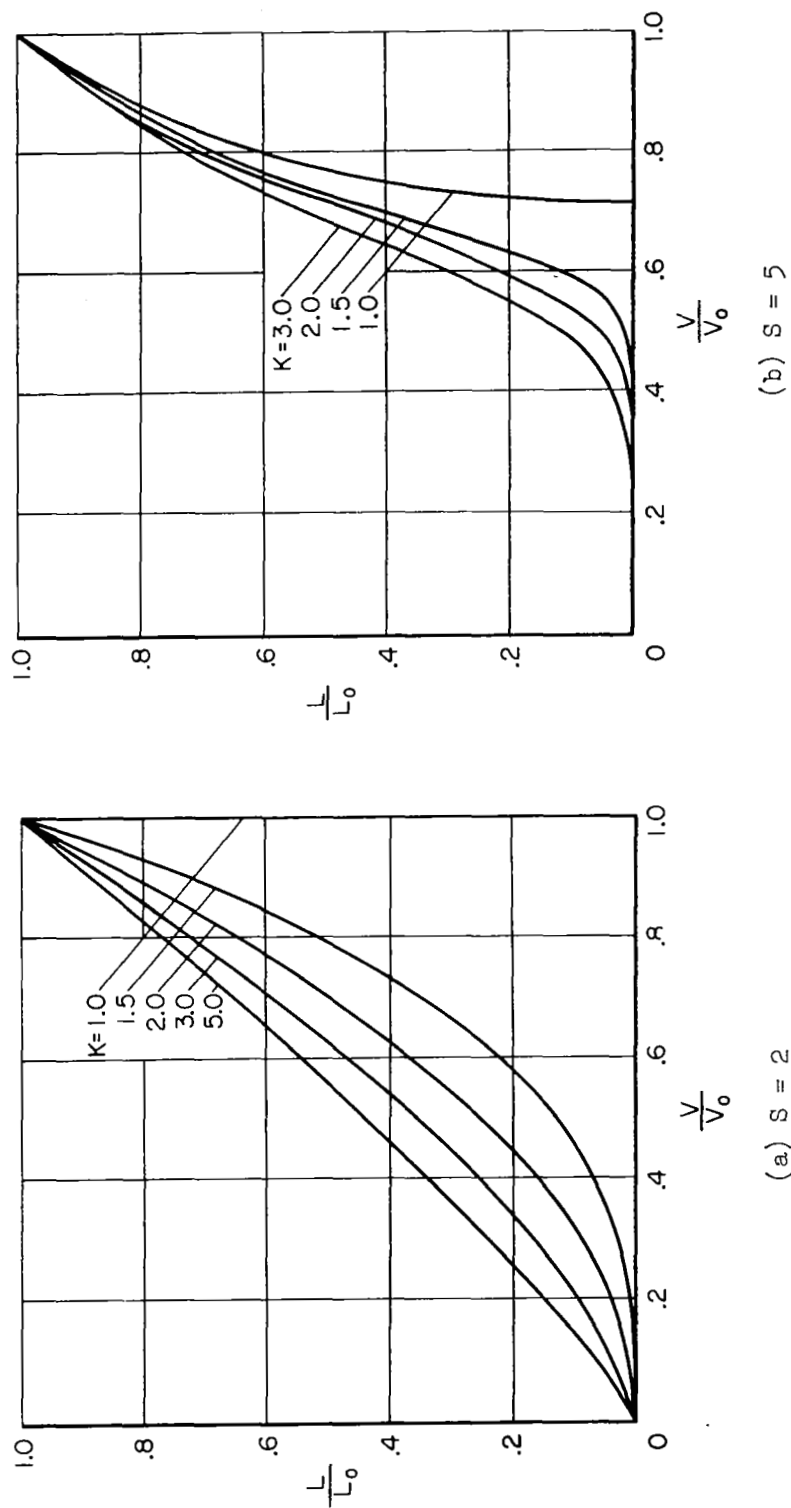
FIGURE C-10. LEAD ANGLE VERSUS VELOCITY, $L_0 = \dfrac{\pi}{2}$

(a) S = 2

(b) S = 5
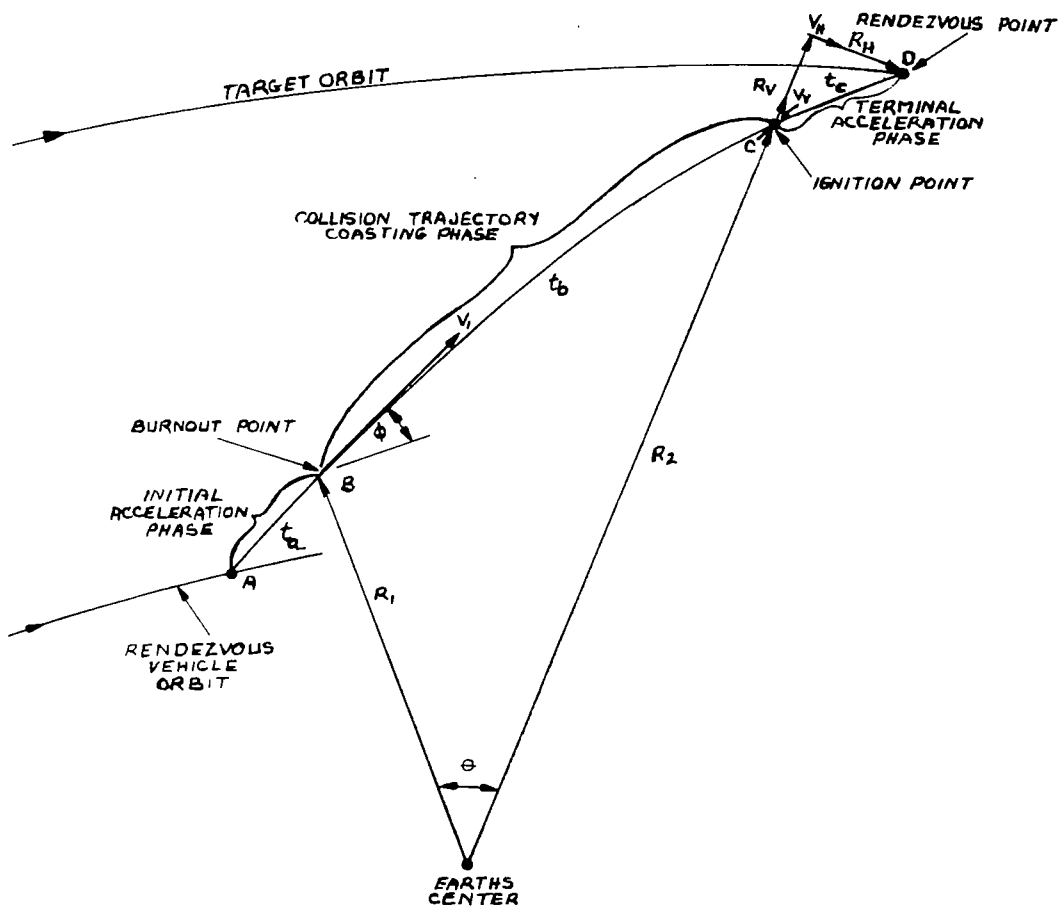
FIGURE C-II. GEOMETRY FOR A TYPICAL TWO IMPULSE
RENDEZVOUS MANEUVER

# APPENDIX D

## HEAD ATTITUDE SENSING METHODS FOR
## HELMET MOUNTED VIEWING SYSTEMS

Space and weight limitations in spacecraft may severely restrict the size of a display optical system. Therefore, one method which has been considered for display is a helmet mounted optical system which includes a small CRT, lenses, and a beam splitter in front of one eye. Such a system has the capability of large fields of view due to the proximity of the collimating element to the pilot's eye. It is necessary, however, to sense head attitude for the display of images which must be located accurately with respect to the real world.

Previous Sperry work in the head position sensing area has been reviewed and extended for application to the spacecraft display requirement. The methods considered cover mechanical connection between the helmet and airframe, as well as several optical and magnetic coupling systems. These systems are summarized in table D-1.

## D-1. MECHANICAL SENSING SYSTEMS

One approach to head attitude sensing is the use of a mechanical attachment between the astronaut's helmet and the airframe. By adapting synchros or variable resistors to this system, any head motion can be detected. Such a system has been fabricated and laboratory and flight tested on previous Sperry programs. The mechanical system consists of a primary and secondary set of angular sensing devices (synchros or potentiometers) connected by a rigid sliding rod (see figure D-1). The primary angular sensors are referenced to the sliding rod and the secondary set is referenced to a fixed body such as an aircraft structure. Both sets of sensors measure angular motions in elevation and azimuth with respect to their references. The primary and secondary sensors are electrically wired together so that any angular motions between them are algebraically added. Translation of the primary sensors with respect to the secondary sensors results in zero total angular displacement. In operation, the primary set of sensors is connected to the operator's helmet. Thus any motion of the operator's head, in azimuth and in elevation, is transposed as an angular signal which may be used in a display computer. The mechanical system has the following physical parameters:

# TABLE D-1

## COMPARISON OF HEAD ATTITUDE SENSING TECHNIQUES

| Technique | Location of Sensors | Types of Sensors | Weight on Helmet | Allowable Head Motion | Problem Areas |
|---|---|---|---|---|---|
| 1. Mechanical | Helmet and airframe | Synchros, potentiometers or resolvers | Minimized by use of counter balance (nil) | -Pitch ± 70° <br> -Yaw ± 180° <br> -Roll ± 90° <br> -Translation-within 10-inch radius sphere | -Direct connection between operator's head and airframe |
| 2a. Optical-refraction | Helmet and airframe | -I.R. source (0.8 micron wavelength) <br> -Refractive | Minimum of one-half pound | Pitch and yaw maximum is determined by critical angle of refractive material (approximately 42°) | -Vibration between components <br> -Limited head motion <br> -Ambient light interference |
| 2b. Optical-fixed mirror reflection | Helmet and airframe | -Reflective <br> -Photo cells <br> -Motor driven | One pound | -Pitch ± 70° <br> -Yaw ± 180° <br> -Translation limited to size of reflector | -Ambient light interference <br> -Weight <br> -Boresighting |
| 2c. Optical-moving mirror reflection | Helmet and airframe | -Reflective <br> -Photo cells <br> -Motor driven | One-half pound | Same as 2b | Same as 2b |

TABLE D-1

COMPARISON OF HEAD ATTITUDE SENSING TECHNIQUES (Cont)

| Technique | Location of Sensors | Type of Sensors | Weight on Helmet | Allowable Head Motion | Problem Areas |
|---|---|---|---|---|---|
| 3. Magnetic | Helmet and airframe | Hall generators | Nil | -Pitch ± 90°<br>-Yaw ● 90°<br>-Roll ± 90°<br>-Translation - must stay within magnetic field | -Power required and size of components required for creating magnetic field<br>-Cross coupling effects among pitch, yaw and roll |

| Elements | Weight (pounds) | Volume (cu in.) | Power (watts) |
|---|---|---|---|
| Primary Sensors (Synchros) | 0.2 | 2 | 2.5 |
| Secondary Sensors | 0.25 | 2 | 2.5 |
| Sliding Rod | 1.50 | 5 | 0 |

The system is constructed with all the weight supported at the secondary sensor refrerence in the airframe. The results of laboratory and flight tests indicate that the angular sensing accuracy is directly dependent upon the angular sensor component accuracy. The overall dimensions of the angular sensing device are 20 inches long by 2 inches square. This size allows the operator complete head motion in all directions. The particular configuration shown in figure D-1 was designed and built for use in helicopter armament control. While this system works well for its intended use, a more advantageous system would be obtained if the connection between the airframe and the pilot could be removed. This can be done by making use of optical or magnetic sensors.

## D-2. OPTICAL SENSING SYSTEMS

The state of the art of optics offers one of the best means for obtaining a high degree of accuracy in a helmet attitude sensing system. In the following paragraphs some of the optical techniques which may be employed are reviewed, and a specific system is recommended.

### A. REFRACTION TECHNIQUE

In the refraction technique, as the pilot moves his head in azimuth and in elevation, the refractive material displaces the light beam from a point in line with the collimated light source (see figure D-2). The nulling refractor is driven to an equal and opposite angle to restore the light beam to the null position. The angular position of the head is then obtained from synchro control transmitters on the nulling refractor servo shafts.

It is important that a refractive medium be selected which has a high refractive index per unit weight, as the additional weight on the operator's helmet must be kept to a minimum. Infra-red light is used to maximize the index of refraction and thus system angular sensitivity. This system limits angular sensing to the critical angle of the refraction material (approximately 42 degrees).

## B. FIXED MIRROR REFLECTION TECHNIQUE

In the fixed mirror reflection technique, the angular motion of the operator's head causes the collimated light beam to reflect from the mirror and strike the detector off the center null position (see figure D-3). This error is used to pulse a stepping motor in the proper direction to restore the light beam to the null position. A count of the number of pulses needed to restore the null is a measure of the angular movement of the operator's head. This is essentially the same as having an autocollimator mounted to the helmet.

One of the major problems with the optical systems is the effect of cross coupling amoung pitch, yaw and roll. This problem also exists in the mechanical and magnetic systems; however, in the mechanical system, roll freedom can be mechanical prohibited, which greatly reduces the cross coupling error, or roll error can be detected directly by a synchro or potentiometer and used for cross-coupling error correction in a computer.

In the above optical systems, cross-coupling error can be practically eliminated if the roll angle is detected in some usable form. One possible means of obtaining the roll angle is by using polarized light and light sensitive elements as shown in figure D-3. A linear polarizer is placed over the lens of the light source, polarizing the emerging light in one plane. The mirror is half-silvered (50 percent transmission and 50 percent reflection) so that half the polarized light is returned to the light detector for control of azimuth and elevation, while the other half passes through the mirror. Located behind the mirror is a second plane polarizer which is oriented 45 degrees out of phase with the polarizer over the light source. A mosaic of light sensitive elements is located behind the second polarizer. In operation, the polarized light that passes through the mirror will also pass through the second polarizer to the light sensors. The amount of light reaching the light sensors is dependent upon the orientation of the polarizers with respect to each other, and the output signal of the light sensors varies as the light intensity changes. All the elements, the light source, mirror, polarizers and light sensors, are mounted parallel to each other and perpendicular to the light path. In the boresight null position, the polarizers are crossed at 45 degrees to give a reference for determining the direction of roll. For instance, if the light sensors indicate an increase in intensity above the null intensity, the roll angle is clockwise, and if the sensors indicate a decrease in intensity, the roll angle is counterclockwise.

Using suitable sensors, this system could detect operator head roll angles of ± 30 degrees with an accuracy of 3 to 4 mils. The signals from the sensors are used in a computer for correcting cross-coupling errors, and are used directly in the display computer for image roll stabilization.

## C. MOVING MIRROR REFLECTION TECHNIQUE

This system utilizes a flat mirror on a gimbaled mount on the helmet with the light source and angular sensors fixed to the airframe (see figure D-4). As the pilot turns his head, the system drives the mirror to the null position, thus indicating angular head motion. As in the fixed mirror system, translatory head motion is limited to the size of the mirror. Roll angle is detected through the use of polarization techniques, as in the fixed mirror system. In both systems, one of the polarizers may be rotated to a null intensity to detect roll angle to avoid nonlinear roll calibrations required if the light sensor output is used directly.

## D-3  MAGNETIC SENSING SYSTEMS

The general requirements for a magnetic pickoff system are:

- Uniform Magnetic Field

- Magnetic Field Sensors.

The existing Earth's magnetic field is relatively uniform and readily available. However, the flux density is low and requires large and heavy magnetic field amplifiers. It is also subject to local variations and is influenced by ferrous materials which may restrict the location of sensing components. It cannot be utilized for space applications. These problems may be overcome by using an induced magnetic field. Some sources of induced magnetic field are permanent magnets, electro-magnets or electric coils.

Specifically, using Helmholtz coils, a uniform magnetic field can be obtained and its direction, magnitude and frequency can be controlled. Choice of frequency makes possible the elimination of the effects of existing fields and magnetic noise. For these reasons an induced magnetic field is ideal for our requirements.

The sensors which may be used are called Hall-effect generators, or simply Hall generators. This device is a solid-state element which provides an output voltage $V_H$ proportional to its orientation in a magnetic field and which varies sinusoidally with the angle between the plane of the Hall generator and the direction of the field (see figure D-5). Angles of azimuth and elevation may therefore be obtained by solving for this angle. In the case shown, $\psi$ or $\theta$ equals 90 degrees and $V_H = K_R$, a constant.

Hall-effect generators are presently being used in devices providing scientific breakthroughs in areas such as solid-state resolvers and synchros, transducers and magnetometers, in a frequency range from dc to megacycles. Hall generators can weigh less than an ounce, have an area much less than a square inch and have a power rating in the milliwatt range. Control current is in the milliamp range, while the Hall voltage can reach 200 millivolts or more.

Magnetic pickoff systems can be classified into three types:

- Primary Closed Loop - In this system, an initial voltage, which is dependent upon the position of a sensor in a magnetic field, is nulled at its source, either by further controlled movement of the sensor or the field. This controlled movement is converted to an angle which is a measurement of the position of the sensor. This method is extremely accurate, while requiring the most simple mathematical solution. However, it requires the most complex system mechanically.

- Secondary Closed Loop - In this system a voltage magnitude from the Hall generator is sent to a servo which is nulled and tracked, thereby converting the voltage to an angle. This system requires moderate mechanical and mathematical functions, although the sensitivity and range are excellent.

- Open Loop System - This system mathematically converts a voltage magnitude directly to an angle. This system offers maximum accuracy, with a maximum of mathematical complexity. Mechanical needs are at a minimum. However, necessary compensations and extensive calibration and maintenance make its use impractical.

It is believed that the secondary closed loop system is the most practical at this time, and this system will now be described in more detail. A set of Helmholtz coils generates a controlled, uniform magnetic field throughout the space through which an operator's helmet moves (see figure D-5). Two Hall generators, one for azimuth, $\psi$, and one for elevation, $\theta$, are mounted on the helmet so that $\psi$ and $\theta$ are zero, and the induced Hall voltage is zero.

If the Hall generators are rotated through an angle in azimuth and elevation, two separate voltages, $V_\psi = K_R \sin\psi - F(\theta)$, and $V_\theta = K_R \sin\theta - F(\psi)$, are obtained. These voltages represent functions of angles which can be solved by using nulling servo techniques.

The input to the $\psi$ nulling servo is $V_\psi = K_R \sin\psi$. As the servo drives to null, resolver feedback will become equal to $-V_\psi$, and a synchro control transmitter will transmit the angle of azimuth to the display computer. Elevation is computed in the same manner.

The advantages of this system are numerous. The pickoff system, excluding the nulling servos, is solid state, lightweight, has no moving parts, and allows the operator unrestricted head motion, maximum safety and comfort. One disadvantage of this system is a cross-coupling effect between $\psi$ and $\theta$. This problem can be overcome by driving the magnetic field or the Hall generators to a null position. A second and major disadavantage for spacecraft application is the fact that the Helmholtz coils are large and require large amounts of energizing power.

## D-4. ACCURACY CONSIDERATIONS

An error analysis for the five head attitude sensing systems is presented in table D-2. The total root-sum-square error ranges from 3.1 mils for the mangetic to 3.6 mils for the moving mirror optical system. These errors are based on a one-speed pickoff system and would be reduced somewhat with a 2-speed system. The errors are shown without roll compensation for the cross-coupling error. The RSS errors would be approximately 0.5 mil less with roll compensation.

## D-5. SUMMARY

Mechanical, optical and magnetic head attitude sensing methods have been described. The mechanical system has been proven in actual use but requires direct connection between the astronaut and the airframe. It is not considered desirable, therefore, if crew movement from the control station is anticipated. However, it would be the cheapest and most readily implemented system for feasibility tests in conjunction with a heads-up/head-mounted display system.

The magnetic system involves too high a weight and power penalty with present state-of-the-art techniques and is therefore not suitable for spacecraft application.

The optical systems offer the best long-term solution to head attitude sensing in spacecraft. They are lightweight and require no direct connection between the astronaut and the air frame. Such systems require further development and test, however, to investigate the actual weight increment to the helmet and overall accuracies operationally attainable.

As the equipment state-of-the-art improves, other sensing techniques may be considered, such as miniature helmet-mounted gyros or highly directional radio signals.

## TABLE D-2

### COMPARISON OF SYSTEM ERRORS

| Source of Error | Type | Angular Contribution to System Error (1$\sigma$) Mils | | | | |
| --- | --- | --- | --- | --- | --- | --- |
| | | Mechanical | Optical 1 (Refraction) | Optical 2 (Fixed Mirror) | Optical 3 (Moving Mirror) | Magnetic |
| Boresight of helmet to pickoff system | Electro-mechanical | ± 3.0 | ± 1.0 | ± 1.5 | ± 1.5 | ± 0.5 |
| Vibration error | Mechanical | ± 1.5 | ± 1.5 | ± 1.0 | ± 1.5 | ± 1.0 |
| Pickoff error | Electro-mechanical | ± 2.0 | ± 0.5 | ± 0.5 | ± 0.5 | ± 0.5 |
| Follow-up servo | Electro-mechanical | ± 2.0 | ± 2.0 | ± 2.0 | ± 2.0 | ± 2.0 |
| Cross-coupling error | Electro-mechanical | ± 1.0 | ± 2.0 | ± 2.0 | ± 2.0 | ± 2.0 |
| Total (RSS) error | | 4.5 | 3.4 | 3.4 | 3.57 | 3.08 |

Note: Use of roll compensation to reduce cross-coupling errors would reduce RSS error approximately 0.5 mil.

Primary
Elevation Pickoff

Primary
Azimuth Pickoff

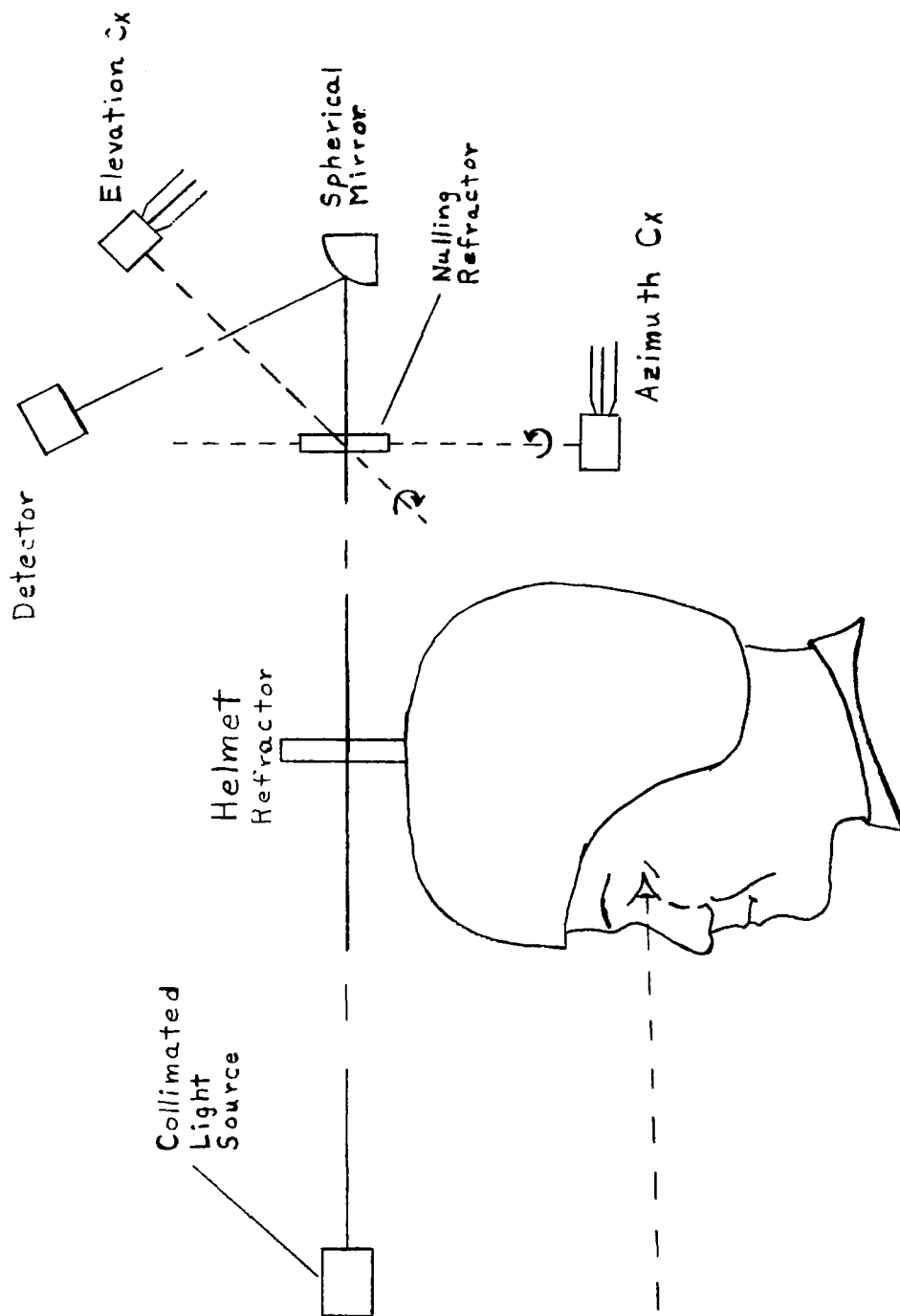Attachment to Helmet

FIGURE D-1
MECHANICAL HEAD SENSING EQUIPMENT
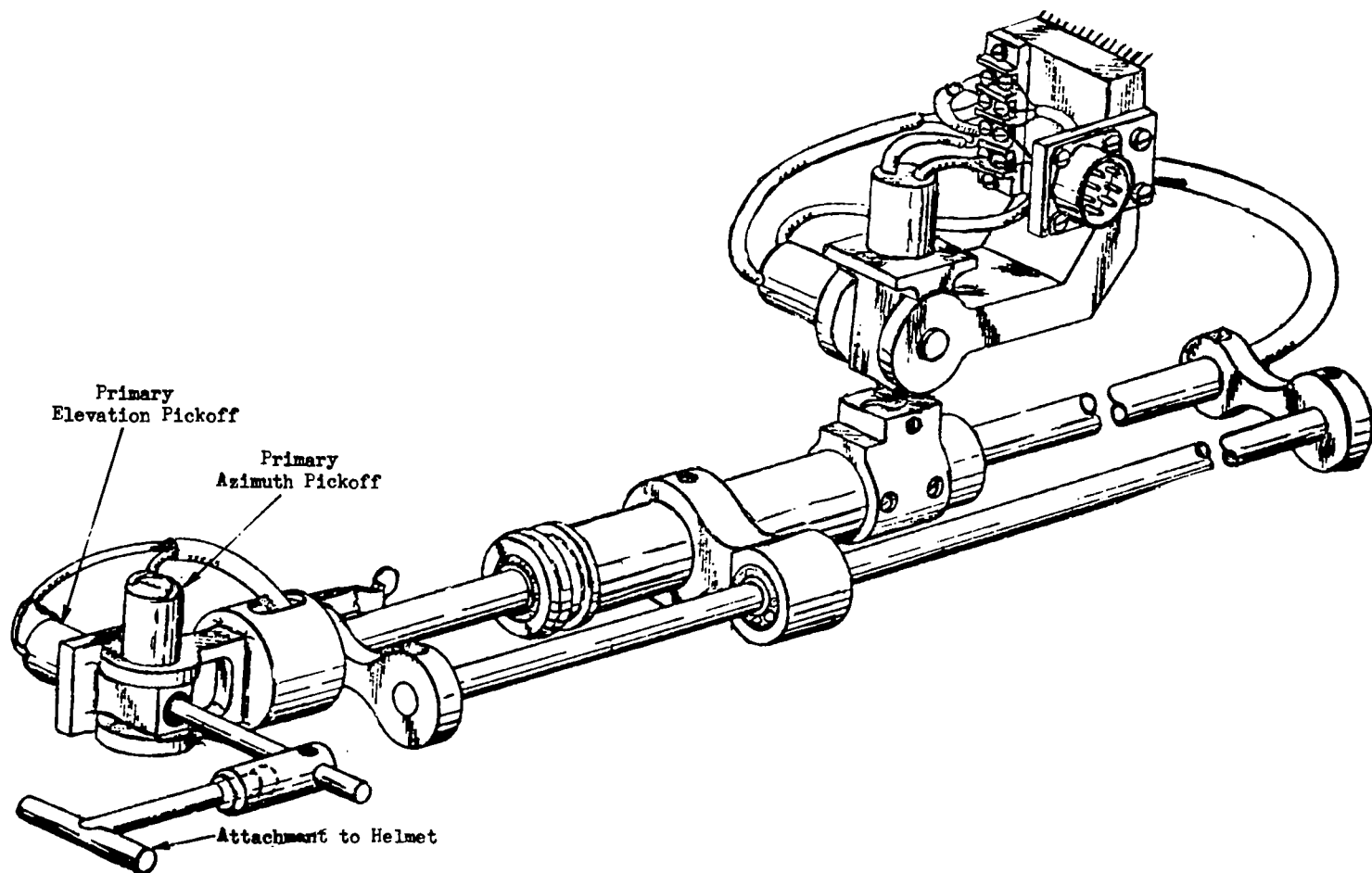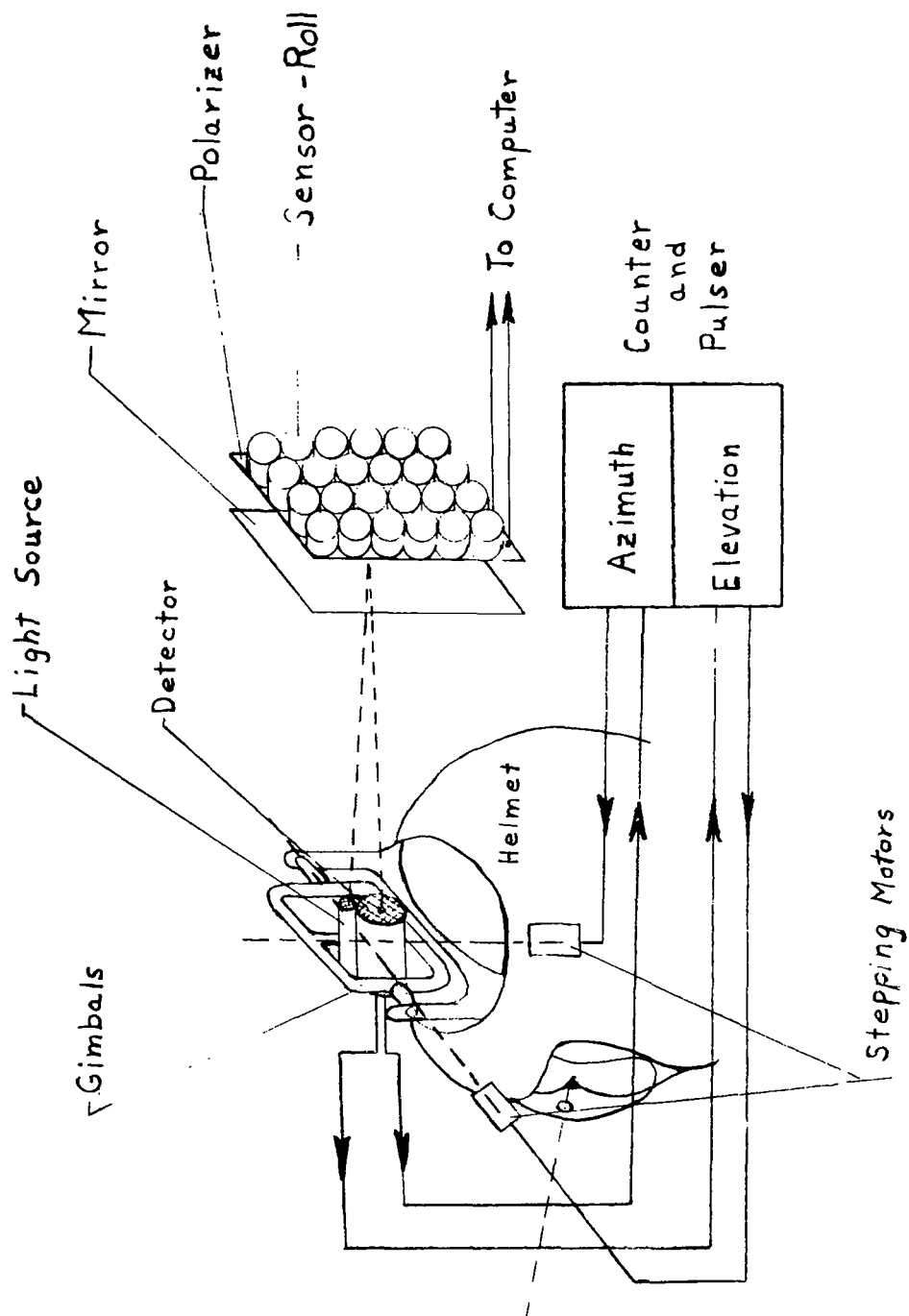
FIGURE D-2. OPTCAL REFRACTION HEAD ATTITUDE SENSING TECHNIQUE

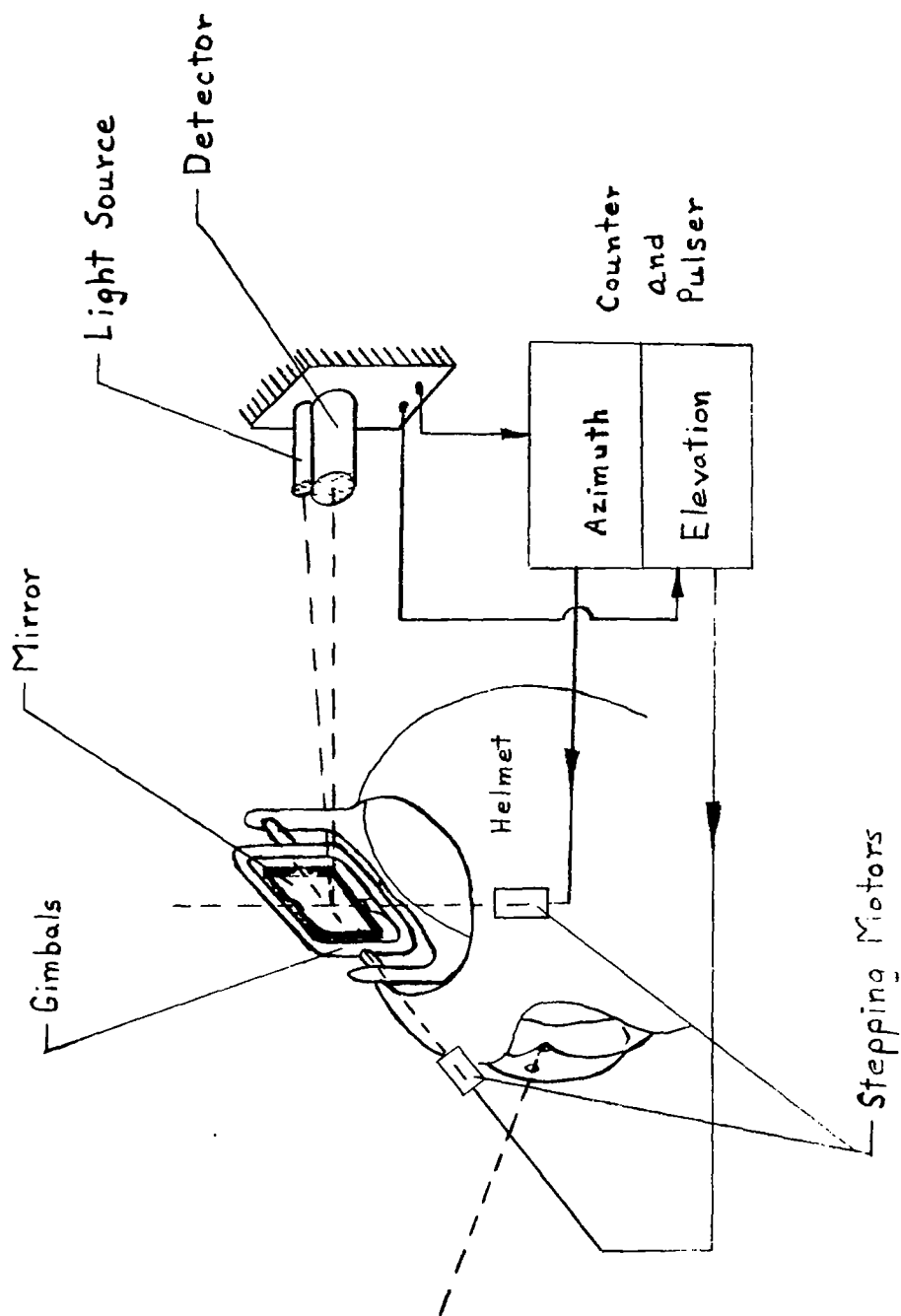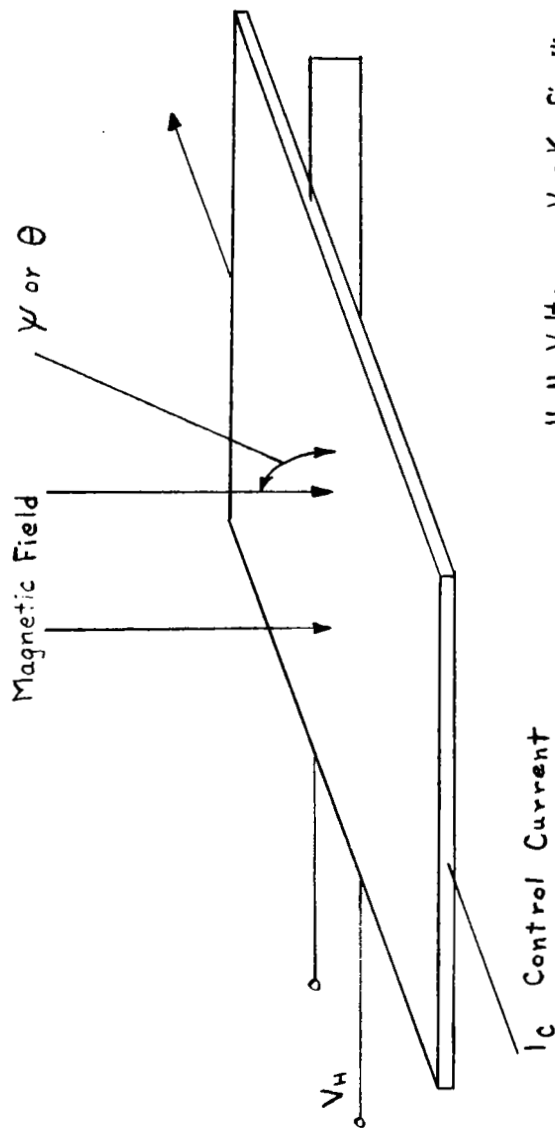FIGURE D-3. FIXED MIRROR REFRACTION TECHNIQUE FOR
SENSING HEAD ATTITUDE

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FIGURE D-4. MOVING MIRROR TECHNIQUE FOR
HEAD ATTITUDE SENSING

Magnetic Field

$\psi$ or $\theta$

$V_H$

$I_c$ Control Current

Hall Voltage = $V_H$ = $K_R$ Sin $\psi$ or $\theta$

$K_R$ = Relative Constant Depending on $I_c$, Flux Density of Field, and Chemical and Physical Characteristics of Hall Generator

$\psi$ = Azimuth Angle

$\theta$ = Elevation Angle

FIGURE D-5. HALL GENERATOR

# APPENDIX E

## RELIABILITY ANALYSIS

### E-1. RELIABILITY

The reliability of the display system can be estimated by means of the product rule. It is assumed that the failure of any electronic part would cause failure of the system and that all failures are independent. It is also assumed that the failure distribution of all parts is exponential; that is, any part has a constant failure rate. Under this assumption, the probability of a part surviving until time t is

$$R_i(t) = 1 - \int_0^t \lambda_i e^{-\lambda_i t} = e^{-\lambda_i t}$$

Since all failures are considered independent, and since the failure of any part in the assembly is assumed to cause failure of the whole assembly, the probability of a failure in an assembly of n parts by time t is

$$R_A(t) = \prod_{i=1}^{n} R_i(t) = \prod_{i=1}^{n} e^{-\lambda_i t} = e^{-\sum_{i=1}^{n} \lambda_i t} = e^{-\lambda_A t}$$

Again, the failure of any assembly is assumed to cause the display system to fail and assembly failures are considered independent. The probability a display system of N assemblies failing is then

$$R(t) = \prod_{i=1}^{N} R_A(t) = \prod_{i=1}^{N} e^{-\sum_{i=1}^{N} \lambda_A t} = e^{-\lambda t}$$

The mean of exponential distribution, or the reliability function, is expressed by

$$\int_0^\infty R(t)\,dt = \int_0^\infty e^{-\lambda t}\,dt = \frac{1}{\lambda}$$

Therefore, the reciprocals of $\lambda_i$, $\lambda_A$ and $\lambda$ may be designated $\Theta_i$, $\Theta_A$ and $\Theta$, and are the mean or expected lives of the part, assembly and system, respectively.

The part failure rates used in these reliability estimates were obtained from Mil Handbook 217 and are shown below.

## TABLE E-1

### PREDICTED PART FAILURE RATES

| Part Type | Failure Rate %/1000 hours |
|---|---|
| Cathode Ray Tube | 1.500 |
| Transistors | 0.078 |
| Resistors | 0.008 |
| Diodes | 0.039 |
| Capacitors | 0.001 |
| Capacitor (Electrolytic) | 0.023 |
| Transformers | 0.100 |
| Connectors | 0.100 |
| Relays | 0.100 |
| Integrated Circuits | 0.05 |
| Motor | 0.20 |

### A. SYSTEM NO. 1

Reliabilities for the three systems described in paragraph 3-7 have been established. The estimated failure rate for each assembly of System No. 1 is shown in the table E-2.

## TABLE E-2

### ESTIMATE FAILURE RATES FOR SYSTEM NO. 1 ASSEMBLIES

| Assembly | Failure Rate %/1000 hours |
|---|---|
| Regulated Power Supply (69566) | 1.654 |
| D-C Amplifier (2) (BC-10620) | 3.002 |
| D-C Motors (2) | 0.400 |
| Power Transformer | 0.100 |
| Meter | 0.167 |
| Total | 5.323 |

The estimated mean-time-between-failure ($\Theta$) for System No. 1 is approximately 18, 700 hours.

The reliability or probability of success for a mission time of 120 hours is

$$R(120) = e^{\frac{-120}{18,700}} = 0.9936$$

If, during the five-day mission (120 hours), the use cycle of the element is 50 percent, the reliability or probability of success would be

$$R(60) = e^{\frac{-60}{18,700}} = 0.9968$$

B. SYSTEMS NO. 2 AND 3

The estimated failure rate for each assembly of System No. 2 and System No. 3 is shown in tables E-3 and E-4, respectively.

## TABLE E-3

### ESTIMATED FAILURE RATES FOR SYSTEM NO. 2 ASSEMBLIES

| | Quantity | P/N | Failure Rate %/1000 Hours | | | |
| --- | --- | --- | --- | --- | --- | --- |
| | | | Conventional System | | Microelectronics System | |
| | | | Per Card | Total | Per Card | Total |
| Dual DeMod, Type I | 2 | 69604 | 2.491 | 4.982 | 2.491 | 4.982 |
| Dual Servo Amp. | 1 | 69596 | 1.720 | 1.720 | 1.720 | 1.720 |
| X-Y Pots Driving Amplifiers | 1 | 69519 | 0.823 | 0.823 | 0.823 | 0.823 |
| X-Y Deflection Yoke Amplifiers | 2 | 69523 | 0.974 | 1.948 | 0.561 | 1.122 |
| High Voltage Power Supply | 1 | | 1.644 | 1.644 | 1.644 | 1.644 |
| Flip-Flop and Emitter Followers | 1 | 69630 | 3.243 | 3.243 | 1.566 | 1.566 |
| Timing Blanking and Brightness | 1 | 69609 | 2.450 | 2.450 | 0.986 | 0.986 |
| Pulse Amplifier Switch | 1 | 82070 | 7.560 | 7.560 | 5.245 | 5.245 |
| Servo Amp. Power Supply | 1 | 69551 | 0.936 | 0.936 | 0.936 | 0.936 |
| Regulated Power Supply | 1 | 69566 | 1.654 | 1.654 | 1.654 | 1.654 |
| ±25 Volt, High Current Power Supply | 2 | 69568 | 1.233 | 2.466 | 1.233 | 2.466 |
| All Attitude Switch and Pat. Gen. | 2 | 69668 | 4.141 | 8.282 | 4.141 | 8.282 |
| Power Supply Transformer | 1 | 69599 | 0.403 | 0.403 | 0.403 | 0.403 |
| Power Supply Transformer | 1 | 69600 | 0.256 | 0.256 | 0.256 | 0.256 |

ESTIMATED FAILURE RATES FOR SYSTEM NO. 2 ASSEMBLIES (Cont)

| | Quantity | P/N | Failure Rate %/1000 Hours | | | |
|---|---|---|---|---|---|---|
| | | | Conventional System | | Microelectronics System | |
| | | | Per Card | Total | Per Card | Total |
| 3.2 kc Generator | 1 | 69619 | 1.971 | 1.971 | 1.971 | 1.971 |
| 6.4 kc Generator | 1 | 69621 | 1.636 | 1.636 | 1.636 | 1.636 |
| Generator Amp. and Crossover Network | 1 | 69625 | 1.470 | 1.470 | 1.198 | 1.198 |
| 32 Bit "AND GATE" Amplifier | 1 | 82071 | 10.240 | 10.240 | 3.300 | 3.300 |
| 64 Bit "AND GATE" Amplifier | 1 | 69634 | 7.040 | 7.040 | 2.300 | 2.300 |
| Pattern Generator Hud. | 1 | 69665 | 1.520 | 1.520 | 1.520 | 1.520 |
| All Attitude Pat. Gen. No. 1 | 2 | 69674 | 2.361 | 4.722 | 1.583 | 3.166 |
| All Attitude Pat. Gen. No. 2 | 2 | 69676 | 2.969 | 5.838 | 1.883 | 3.766 |
| Magnetic Multiplier Assembly | 2 | 69658 | 1.169 | 2.338 | 0.949 | 1.898 |
| 3.2 kc Voltage Regulator | 1 | 69683 | 1.130 | 1.130 | 1.130 | 1.130 |
| Pulse Amplifier and Switch No. 2 | 2 | 69689 | 5.488 | 10.976 | 3.529 | 7.058 |
| Pulse Amplifier and Switch No. 3 | 1 | 69691 | 3.504 | 3.504 | 3.504 | 3.504 |
| Hud. Relay Card No. 2 | 1 | 69697 | 4.035 | 4.035 | 4.035 | 4.035 |
| Hud. Matrix No. 1 | 1 | 82077 | 5.772 | 5.772 | 5.772 | 5.772 |
| Hud. Matrix No. 2 | 1 | 82076 | 1.405 | 1.405 | 1.405 | 1.405 |
| Test Card No. 3 | 1 | 69703 | 1.853 | 1.853 | 1.853 | 1.853 |

## TABLE E-3

### ESTIMATED FAILURE RATES FOR SYSTEM NO. 2 ASSEMBLIES (Cont)

| | Quantity | P/N | Failure Rate %/1000 Hours | | | |
| | | | Conventional System | | Microelectronics System | |
| | | | Per Card | Total | Per Card | Total |
|---|---|---|---|---|---|---|
| Trimpots | 20 | | 0.100 | 2.000 | 0.100 | 2.000 |
| Schmitt Trigger Comparators | 15 | | 0.244 | 3.660 | 0.143 | 2.150 |
| AND/OR Gates | 34 | | 0.203 | 6.902 | 0.050 | 1.700 |
| Full Wave Rectifier | 2 | | 0.186 | 0.372 | 0.186 | 0.372 |
| Dial Pots | 10 | | 0.100 | 1.000 | 0.100 | 1.000 |
| Multivibrators | 3 | | 0.300 | 0.900 | 0.300 | 0.900 |
| Pitch Servo | 1 | 69624 | 2.300 | 2.300 | 2.300 | 2.300 |
| Roll Servo | 1 | 66969 | 1.000 | 1.000 | 1.000 | 1.000 |
| Yaw Servo | 1 | | 2.300 | 2.300 | 2.300 | 2.300 |
| CRT | 1 | | 1.500 | 1.500 | 1.500 | 1.500 |
| Yoke | 1 | | 0.100 | 0.100 | 0.100 | 0.100 |
| TOTALS - All CRT Display | | | | 125.851 | | 97.819 |
| - Two-Color Display (Addition of System No. 1) | | | | 131.174 | | 103.142 |

## TABLE E-3

## ESTIMATED FAILURE RATES FOR SYSTEM NO. 2 ASSEMBLIES (Cont)

| | Conventional System | | Microelectronic System | |
|---|---|---|---|---|
| | All CRT | Two Color | All CRT | Two Color |
| Estimated mean-time-between-failure $\Theta$ - Hours | 795 | 762 | 1022 | 970 |
| Probability of success for a 5-day (120 hour) mission (reliability) is $R(120) = \exp \frac{-120}{\Theta}$ | 0.8607 | 0.8543 | 0.8891 | 0.8840 |
| Probability of success for a 5-day (120 hour) mission with use cycle of 50 percent is $R(60) = \exp \frac{-60}{\Theta}$ | 0.9278 | 0.9240 | 0.9427 | 0.9399 |

# TABLE E-4

## ESTIMATED FAILURE RATES FOR SYSTEM NO. 3 * ASSEMBLIES

|  | Quantity | Failure Rate %/1000 Hours |
|---|---|---|
| Magnetic Mods | 8 | 4.676 |
| Crossover Detectors | 13 | 4.576 |
| Dot Transformers | 5 | 0.500 |
| Trimpots | 10 | 1.000 |
| Capacitors (tantalum) | 11 | 0.253 |
| "AND" Gates | 20 | 4.420 |
| Gates | 20 | 3.500 |
| D-C Amplifiers | 12 | 10.320 |
| Mag Mod | 14 | 8.183 |
| Switches | 34 | 7.560 |
| Landing Area Computer |  | 44.988 |

*System No. 3 comprises the Landing Area Computer and System No. 2.

|  | Failure Rate %/1000 Hours | |
|---|---|---|
|  | All CRT | Two Color |
| System No. 2 | 125.851 | 131.174 |
| Landing Area Computer | 44.988 | 44.988 |
| System No. 3 Failure Rate | 170.839 | 176.162 |
| Mean-Time-Between-Failure (Θ) - Hours | 585 | 568 |

Probability of success (reliability) for a 5-day (120 hour) mission is:

$$R(120) = e^{\frac{-120}{\Theta}} \qquad = \qquad 0.8146 \qquad 0.8092$$

Probability of success for a 120 hour mission with a use cycle of 50% is:

$$R(60) = e^{\frac{-60}{\Theta}} \qquad = \qquad 0.9026 \qquad 0.8994$$

## E-2. REDUNDANCY CONSIDERATIONS

Employing redundant elements is a technique for improving reliability. The value of redundancy becomes evident from the following analysis.

Assuming: probability of success of an element = $p_1$

probability of failure of same element = $q_1$

probability of success of remainder of system = $p_0$

probability of failure of remainder of system = $q_0$

then $\qquad p_1 + q_1 = 1$ ; $p_0 + q_0 = 1$

and $\qquad (p_1 + q_1)(p_0 + q_0) = 1$

therefore,

$$p_1 p_0 + q_1 p_0 + p_1 q_0 + q_1 q_0 = 1$$

Of these terms, $p_1 p_0$ is the only one that denotes overall success. Now if redundancy is employed for the element, using a second element, $p'_1 + q'_1 = 1$, then the following product may be formed:

$$(p_1 + q_1)(p_1' + q_1') = 1$$

which equals

$$p_1 p_1' + p_1 q_1' + p_1' q_1 + q_1' q_1 = 1$$

If successful operation is assured by either the element or its redundant element operating properly, the only term denoting failure is $q_1' q_1$. Returning to $p_1 p_0$, the overall success probability, its form is now altered to $(1 - q_1)(p_0) = p_0 - q_1 p_0$. In place of $q_1$ now substitute $q'_1 q_1$, the new effective probability of failure, to get

$$\text{Probability of Success} = p_0 - (q'_1 q_1) p_0$$
(with redundancy)

Note that $(q'_1 q_1)$ is much smaller than $q_1$ since $q_1' \ll 1$.   As an example, if all p's are 0.9 and all q's are 0.1, then

Probability of Success Without Redundancy $= p_1 p_o = (0.9)(0.9) = 0.81$

and

Probability of Success With Redundancy $= 0.9 - (0.1)(0.1)(0.9)$

$$0.9 - 0.009 \approx 0.9$$

The 0.009 probability is insignificant, thus indicating that redundancy has removed the probability of failure of the element in question as a serious factor in the overall reliability.   These relationships apply to the systems in this report as follows:  Consider   System No. 3 on a 5-day, 120-hour,  mission with all CRT display; the probability of success = 0.8146. This is equal to the product of the probability of success of System No. 2 (0.8891) and the probability of success of the Landing Area Computer (as per the $p_1 p_o$ term),  or $(0.8146) = (0.8891)$ x (probability of success of Landing Area Computer); therefore,

Probability of Success of Landing Area Computer $= \dfrac{0.8146}{0.8891} = 0.917 = p_{LAC}$

Now,  if the Landing Area Computer is made redundant,  the resultant probability of success would be expected to approach 0.8891 (System No. 2).  This can be shown as follows:

$0.8146 = p_{Sys.\ No.\ 2} - (q_{LAC})(p_{Sys.\ No.\ 2})$

$\qquad = (0.8891) - (1 - 0.917)(0.8891)$

$\qquad = (0.8891) - (0.083)(0.8891) =$ System No. 3 success probability
$\qquad\qquad\qquad\qquad\qquad\qquad\qquad\qquad$ without redundancy

Now,  in place of $q_{LAC}$ substitute $(q_{LAC})^2$,  the new effective failure probability for the landing area computer.   Then,  the probability of success with redundancy

$\qquad = 0.8891 - (1 - 0.917)^2 (0.8891)$

$\qquad = (0.8891) - (0.0069)(0.8891)$

$\qquad \approx 0.8891$

Similarly, it is evident that if the Electronics Box is made redundant, the resultant probability of success approaches 0.917.

The above analysis indicates that redundancies are very effective in improving reliability, but these are predicated on several conditions:

1.  Means must be provided to indicate when a malfunction has occurred.

2.  Provisions must be included for engaging the redundant devices when required.

3.  The additional materials required to effect 1 and 2 above are not prohibitive and do not lower the resultant reliability below that without redundancy.

If major subassemblies are the only elements that can be made redundant, it is a condition that approaches having two systems side by side with freedom of choice in interconnection of units. This results in a very high order of reliability but doubles the hardware size and weight.

# APPENDIX F

## DERIVATION OF EQUATIONS USED IN OPTICAL NOMOGRAPH
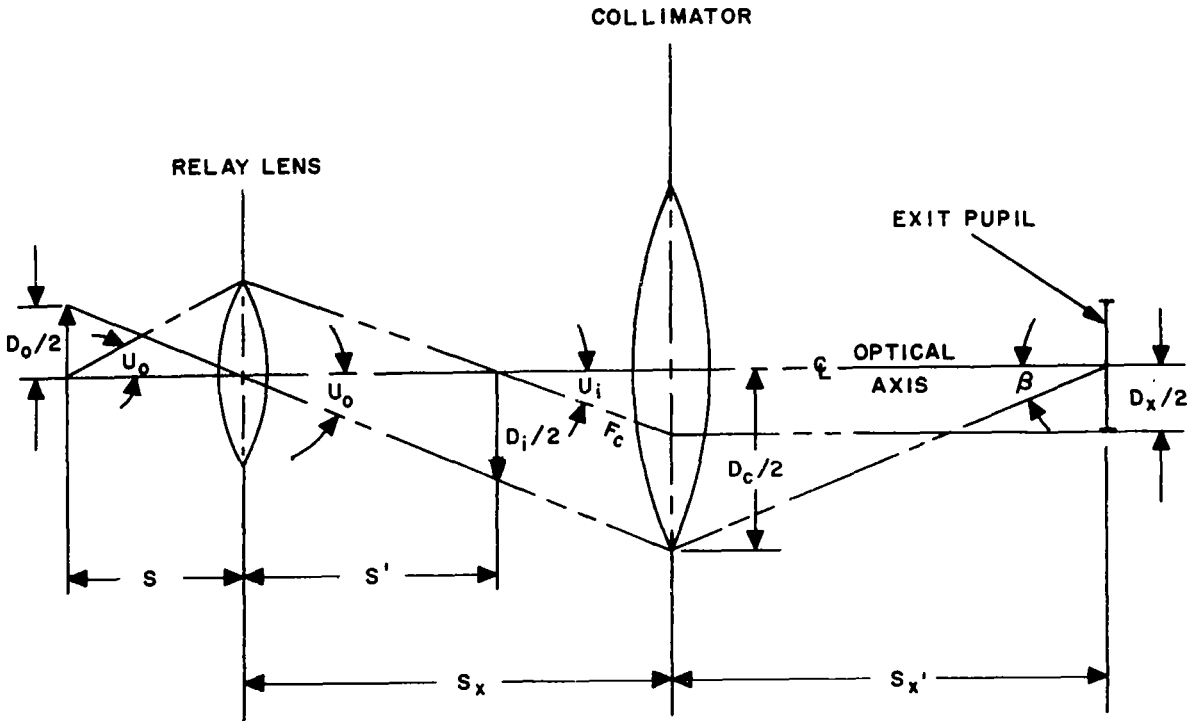


The above diagram shows the optical relationship used in the derivation of the equation for the optical nomograph (figure 3-6.9). Two rays are shown, one a chief ray leaving the maximum object height $D_o/2$ and entering the center of the entrance pupil, and again reaching the axis at the center of the exit pupil. The other ray is an axial ray entering the relay lens pupil at the maximum aperture.

The optical invariant, from the Military Standardization Handbook MIL-HDBK-141 in a form which includes the sine condition for well-corrected systems, is

$$\phi = \frac{D_o}{2} \sin u_o = \frac{D_x}{2} \sin \beta \qquad \text{(F-1)}$$

where,

$\dfrac{D_o}{2}$ = the radial distance of a point on the CRT from the center, taken to be the maximum distance used

$u_o$ = the angle that a rim ray makes with the axis as it leaves the center of the CRT

$\dfrac{D_x}{2}$ = the radius of the exit pupil

$\beta$ = the half-angle field-of-view

The relay lens subtends a cone from the CRT called the object cone of the relay lens and a cone from the projected image of the CRT which is called the image cone of the relay lens. The relay lens is operating at finite conjugates. The special case of an image cone when the lens is operating at an infinite object distance is described as the f/number of the lens. The relationship between object cone, image cone, and f/number is as follows:

$$1/f_o + 1/f_i = 1/f \qquad\qquad\qquad (F-2)$$

where,

$f_o$ = object cone f/No. = $1/(2 \sin u_o)$ $\qquad\qquad$ (F-3)

$f_i$ = image cone f/No. = $1/(2 \sin u_i)$ $\qquad\qquad$ (F-4)

$f_R$ = f/number of relay lens

$u_i$ = slope angle of rim ray as it meets the axis to form a real image

From equation (F-3),

$$\sin u_o = 1/(2 f_o) \qquad\qquad\qquad (F-5)$$

Substituting in equation (F-1),

$$\frac{D_o}{2 \sin \beta} = D_x f_o = Q \qquad\qquad\qquad (F-6)$$

where Q is a parameter to be defined later. Equation (F-6) is the primary equation used to construct the nomograph, and indeed, since it embodies the optical invariant for the system, completely defines the optical system characteristics.

Solving equation (F-2) for $f_R$,

$$f_R = \frac{f_o f_i}{f_o + f_i} = \frac{M_R f_o}{M_R + 1} \qquad (F-7)$$

since

$$M_R = f_i/f_o = \frac{S'}{S} \qquad \text{(the magnification of the} \qquad (F-8)$$
$$\text{CRT performed by the}$$
$$\text{relay lens)}$$

From (F-8), (F-3) and (F-4),

$$M_R = \frac{\sin u_0}{\sin u_i} \qquad (F-9)$$

From the figure, it is seen that

$$\sin u_i = \frac{D_x}{2F_c} \qquad (F-10)$$

where $F_c$ is the focal length of the collimating lens.

From (F-9) and (F-3),

$$\sin u_o = \frac{M_R D_x}{2 F_c} = 1/(2f_o) \qquad (F11)$$

$$D_x f_o = \frac{F_c}{M_R} = Q, \text{ from (F-6)} \qquad (F-12)$$

Thus, it is seen that the parameter Q, found by dividing the object height $D_o/2$ by the sine of the half-angle field-of-view, is the ratio of the focal length of the collimator to the magnification performed by the relay lens.

The pupil magnification performed by the collimating element is, by definition, the ratio of the unit pupil diameter to the diameter of the relay lens aperture:

$$M_P = \frac{D_x}{D_R} = \frac{S_x{}'}{S_x} \qquad (F-13)$$

The remaining relation is that between the focal length of the relay lens, $F_R$, and the relay magnification, $M_R$. From equation (F-13), $D_R M_P = D_x$ and $f_R D_R M_P = D_x f_R$ since $f_R = \dfrac{F_R}{D_R}$, where $F_R$ is the focal length of the relay lens and $D_R$ is the diameter of the relay lens,

$$F_R M_P = D_x f_R \qquad\qquad (F\text{-}14)$$

From equations (F-6) and (F-7),

$$Q = D_x f_o = D_x \frac{(M_R + 1)}{M_R} f_R$$

$$F_R M_P = \frac{Q}{(M_R + 1)/M_R} \qquad\qquad (F\text{-}15)$$

Thus, on the nomograph, the product $F_R M_P$ may be found either by equation (F-14) or by equation (F-15).

The nomograph is constructed with ten scales. Scales 1 and 8 are used to relate the CRT diameter to the total field-of-view, determining the parameter Q for the system. About the point on Q thus determined, the pupil diameter and object cone of the relay lens are determined on scales 4 and 7. From the $f_o$/number on scale 7 and the relay magnification on scale 10, the f/number of the relay lens is determined on scale 9. From the relay magnification on scale 10, a line back to Q on scale 5 determines the product $F_R M_P$ on scale 6. The same product $F_R M_P$ could also be found by a line from $f_R$ on scale 9 to the pupil diameter, $D_x$, on scale 4.

Using scales 2 and 4 determines the diameter of the relay lens on scale 3.

By adding more scales, it would be possible to calculate the collimator focal length directly, and also the relay lens focal length directly. The resulting chart complication is not warranted for the simple computations needed.